~~CONFIDENTIAL~~

CCN

X63-14172

APOLLO RENDEZVOUS SIMULATOR STUDY

CONTRACT NASw -413(HS-625)

CLASSIFICATION CHANGE

TO - UNCLASSIFIED ~~CONFIDENTIAL~~

By authority of T.D. No. 74-235 ~~174-235~~

Changed by L. Shirley Date 3/22/74

TECHNICAL DATA REPORT

VOLUME VI

VEHICLE ANALYSIS — ENVIRONMENTAL CONTROL SYSTEM

12 JULY 1962

"Available to U.S. Government Agencies and U. S. Government Contractors Only."

REPORT NO. 324.4

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP
BOX 6267 DALLAS 22, TEXAS

N74-73841

(NASA-CR-138065) APOLLO RENDEZVOUS SIMULATOR STUDY. VOLUME 6: VEHICLE ANALYSIS - ENVIRONMENTAL CONTROL SYSTEM Technical Data Report (Chance Vought Corp.) 152 p

Unclas
00/99 32673

EXCLUDED FROM THE GDS

Report No. 324.4
Dated:
12 July 1962

VOLUME VI

TECHNICAL DATA REPORT

VEHICLE ANALYSIS - ENVIRONMENTAL CONTROL SYSTEM

Apollo Rendezvous Simulator Study

Contract NASw - 413 (HS-625)

Submitted By:

Astronautics Division
Chance Vought Corp
P.O. Box 6267 · Dallas 22, Texas

to

National Aeronautics and Space Administration

Prepared By:

G B Whisenhunt

G. B. Whisenhunt
Power & Environment

Checked By:

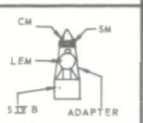
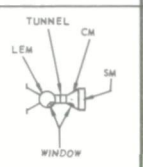
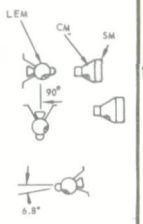
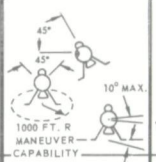
W C Trent

W. C. Trent
Manager
Power & Environment

Approved By:

E. V. Marshall

E. V. Marshall
Program Manager

MISSION PHASE	PHASE IS TERMINATED BY	EVENT SEQUENCE	LEM ORIENTATION	TIME EVENT	TIME CUM (hrs)	ALTITUDE	DELTA V (Ideal) fps	LEM WEIGHT (Earth lbs)
EARTH LAUNCH & PARKING ORBIT	BURNOUT AT TRANSLUNAR INJECTION	● BOOST INTO EARTH PARKING ORBIT ● COAST IN PARKING ORBIT ● TRANSLUNAR INJECTION FROM EARTH PARKING ORBIT		300 sec. 1.5 hrs. 135 sec	.1 1.6 1.7	0 150 NM (earth)	29,760 10,700	29,290
TRANSLUNAR	START OF RETRO INTO LUNAR ORBIT	● COAST ● SEPARATE & JETTISON ADAPTOR ● DOCK COMMAND MODULE – SERVICE MODULE INTO LEM ● SEPARATE SIV B FROM LEM ● PERFORM MIDCOURSE CORRECTION (S) ● COAST TO START OF LUNAR ORBIT INJECTION		72 hrs.	73.7		400	29,290
LUNAR ORBIT & ORBIT TRANSFER	START OF GROSS DECELERATION	● RETRO INTO LUNAR ORBIT ● ESTABLISH ORBIT EPHEMERIS ● TRANSFER ONE CREW MEMBER FROM COMMAND MODULE & INITIATE CHECKOUT OF LEM ● TRANSFER OTHER CREW MEMBER & COMPLETE CHECKOUT ● COAST TO RETRO POINT, OVERPASS LANDING SITE ONCE ● SEPARATE LEM FROM CM ● RETRO TO INITIATE TRANSFER ORBIT ● COAST IN TRANSFER ORBIT ● OBTAIN NAVIGATION DATA		200 sec. 150 min. 26 sec. 29 min.	73.8 76.3 76.3 76.8	59 NM (Lunar) 59 NM	3,000 255	29,894
LUNAR DESCENT	LUNAR TOUCHDOWN	● INITIATE GROSS DECELERATION ● HOVER & LATERAL MOVEMENT. OBSERVE LANDING SITE ● DESCENT TO SURFACE		333 sec. 105 sec.	76.9 77.0	62,000 FT 100 FT 10 FT	5,912 590	15,157
LUNAR SURFACE STAY	START OF LUNAR LAUNCH	● STAGE LANDING GEAR ASSEMBLY ● INSPECTION OF LEM STRUCTURES & SYSTEMS ● EXPLORATION & EXPERIMENTATION ● MAINTENANCE AS REQUIRED ● CHECKOUT & COUNTDOWN		24 hrs.	101.0			11,042
LUNAR ASCENT & ORBIT TRANSFER	COMPLETION OF PLANE CHANGE WITH MAIN ENGINE	● GROSS LAUNCH AND INJECTION HOHMANN TRANSFER ORBIT ● COAST TOWARD ORBIT CONTAINING CM–SM & EXECUTE PLANE CHANGE WITH MAIN ENGINE		228 sec. 43.7 min.	101.1 101.8	50,000 FT.	5,780 25	
LUNAR ORBIT RENDEZVOUS	START OF TRANSEARTH INJECTION	● COAST TO ORBIT CONTAINING CM–SM ● TERMINAL RENDEZVOUS MANEUVERS ● DOCKING ● CREW TRANSFER FROM LEM TO CM ● SEPARATE & ABANDON LEM – COAST TO TRANSEARTH INJECTION POINT		13.3 min. 2.7 min. 14.5 min. 42.8min.	102.0 102.1 102.3 103.0	70,000 FT 59 NM	193 10	5,972 5,248
TRANSEARTH	RE–ENTRY INTO EARTH'S ATMOSPHERE	● TRANSEARTH INJECTION OF COMMAND & SERVICE MODULES ● COAST ● EFFECT MIDCOURSE CORRECTION(S) ● SEPARATE & JETTISON SERVICE MODULE ● COAST PRIOR TO RE–ENTRY	NOT APPLICABLE	200 sec. 72 hrs.	103.1 175.1		3,000 400	N.A.
EARTH RE–ENTRY & LANDING	EARTH LANDING RECOVERY	● RE–ENTRY OF COMMAND MODULE ● EARTH LANDING ● RECOVERY OF COMMAND MODULE & CREW		0.5 hrs.	175.6			

LUNAR ORBIT RENDEZVOUS MISSION OPERATION PLAN – MISSION CHARACTERISTICS

PREFACE

The Lunar Orbit Rendezvous mode for accomplishing the Apollo manned lunar landing has been studied by the Chance Vought Astronautics Division under contract to Office of Systems, Manned Flight, NASA Headquarters. The objective of this study was to make a systematic and thorough analysis of the Lunar Orbit Rendezvous Mission (LOR) with the end products to be (1) A recommended LOR mission, (2) A recommended vehicle design, and (3) A development plan for accomplishing the over-all mission. The study was performed under the title, "Apollo Rendezvous Simulator Study, Contract NASw-413" and is classified CONFIDENTIAL.

The study results are presented in two parts:

Part 1 - SUMMARY REPORT - An over-all summary of the significant results of the study.

Part 2 - A complete TECHNICAL DATA REPORT in eight volumes.

Volume I MISSION SUMMARY AND TRAJECTORY ANALYSIS

Volume II VEHICLE ANALYSIS - DESIGN

Volume III VEHICLE ANALYSIS - PROPULSION

Volume IV VEHICLE ANALYSIS - CONTROLS AND ELECTRONICS

Volume V VEHICLE ANALYSIS - CREW INTEGRATION AND SAFETY

Volume VI VEHICLE ANALYSIS - ENVIRONMENTAL CONTROL SYSTEM

Volume VII VEHICLE ANALYSIS - WEIGHT ANALYSIS

Volume VIII DEVELOPMENT PROGRAM

The study was conducted within the over-all program philosophy and constraints included in the NASA contract statement of work for this study. The principal constraints established in this statement of work are as follows:

- No changes in the Apollo spacecraft design are expected from the result of this study.
- No changes in the Saturn C-5 launch vehicle configuration are expected from the result of this study.

In addition to the contract statement of work, NASA Headquarters defined a series of guidelines for the conduct of the study which were summarized in "Minutes of Lunar Orbit Rendezvous Meeting, April 2-3 1962".

The principal guidelines established by this document are as follows:

- The Lunar Excursion Module (LEM) will have a point landing (±1/2 mile) capability.
- The LEM will have redundant guidance and control for each phase of the lunar maneuvers.
- Both automatic and manual guidance and control systems are to be considered in this redundant capability.
- Radio aids, including use of a beacon and/or transmitter on the lunar surface to provide a completely automatic landing are to be studied.
- The suggested hover capability for the LEM is one minute at 100 ft altitude plus 45 seconds of translation time over the lunar surface. This requirement will be studied further.
- The LEM should include two crew members.
- The LEM should have a pressurized cabin which has a capability for a one week operation.
- Access to the LEM from Apollo during the earth-moon phase should be possible.
- The possibility of keeping the LEM attached to the spacecraft on the return moon-earth phase shall be considered.

In general, the philosophy and guidelines established for the study required an examination of all of the important possibilities and techniques for accomplishing the mission. The resulting recommended mission and vehicle design are therefore more comprehensive than a minimum mission and vehicle. In addition to this recommended vehicle, data are presented showing the effect on vehicle weight of mission and design parameters such as lunar stay time, number of crew members, etc.

This volume presents the LEM Environmental Control System study results and recommendations. Subsystem trade studies, system parametric studies, analysis of special problems, and a description of the operation of the recommended system are included.

TABLE OF CONTENTS

		Page No.
16.0	ENVIRONMENTAL CONTROL SYSTEM	16-1
16.1	INTRODUCTION	16-1
16.2	SUMMARY	16-2
16.3	DISCUSSION	16-5
16.3.1	Life Support Requirements	16-5
16.3.1.1	Cabin Atmosphere	16-5
16.3.1.2	Carbon Dioxide (CO_2) Concentration	16-9
16.3.1.3	Temperatures - Relative Humidity	16-9
16.3.2	Component and Configuration Trade Studies	16-13
16.3.2.1	Oxygen Storage	16-13
16.3.2.2	Carbon Dioxide (CO_2) Removal	16-17
16.3.2.3	Humidity Control	16-17
16.3.2.4	Contaminate Control	16-19
16.3.2.5	Internal Thermal Control	16-19
16.3.2.6	Thermal Control Outside Pressure Shell	16-31
16.3.2.7	Air Lock Trade Study	16-32
16.3.2.8	Translunar Operation	16-34
16.3.2.9	Fuel Cell Environmental Control System Integration	16-34
16.3.2.10	Waste Handling	16-40
16.3.2.11	Reliability Philosophy	16-41
16.3.3	System Parametric Studies	16-45
16.3.4	Analysis of Environmental Control for the Selected Vehicle and Mission	16-49
16.3.4.1	Cabin Leakage	16-49
16.3.4.2	Rocket Plume Expansion	16-49

TABLE OF CONTENTS (Cont'd)

16.3.4.3	Window Thermal Control	16-55
16.3.4.4	Structural Heat Leakage	16-59
16.3.4.5	Thermal Control Loop Analysis	16-70
16.3.4.6	Analog Analysis of Atmosphere Control System	16-80
16.3.5	Recommended Environmental Control System Description	16-87
16.3.5.1	System Definition	16-87
16.3.5.2	ECS Operation	16-87
16.3.5.3	Power Summary	16-109
16.3.5.4	Weight Justification	16-109
16.3.5.5	Changes for 7-day Operation (Lunar Cold Side)	16-117
16.3.5.6	Changes for Lunar Hot Side Operation	16-118
16.3.5.7	Controls and Displays	16-120
16.3.5.8	Ground Checkout Requirements	16-120
16.4.0	DEVELOPMENT PROGRAM	16-127
16.5.0	CONCLUSIONS AND RECOMMENDATIONS	16-129
	BIBLIOGRAPHY	16-130
	APPENDIX	16A-1
	ABSTRACT	

LIST OF ILLUSTRATIONS

Figure No.	Title	Page
16-1	Oxygen-Pressure Effects	16-8
16-2	Phychrometric Chart - Oxygen Atmosphere, 5 psia Total Pressure	16-10
16-3	Phychrometric Chart - Oxygen - Nitrogen Mixture 7.33 psia Total Pressure	16-11
16-4	Comparison of Oxygen Storage System Weights	16-16
16-5	Specific Heat of Heat Transport Fluids	16-26
16-6	Density of Heat Transport Fluids	16-27
16-7	Viscosity of Heat Transport Fluids	16-28
16-8	Comparative Pumping Power for Heat Transport Fluids (Laminar Flow)	16-29
16-9	Comparative Pumping Power for Heat Transport Fluids (Turbulent Flow)	16-30
16-10	Weight Penalty for Ingress and Egress from Pressurized Cabin	16-33
16-11	Air Loss for Pressurized and Unpressurized LEM - Apollo Connecting Tunnel	16-35
16-12	Weight Penalty to Apollo for Translunar Occupancy of LEM (Umbilical Approach)	16-36
16-13	Schematics for Integrated and Non-Integrated Fuel Cell and ECS Cryogenic Storage	16-38
16-14	Schematics for Integrated and Non-Integrated Fuel Cell and ECS Thermal Control Loops	16-39
16-15	ECS Schematic for Space Suit Environment Vehicle	16-46
16-16	Environmental Control System Weight Summary	16-48
16-17	Cabin Atmosphere Leakage	16-50
16-18	LEM Cabin Decompression Time	16-51
16-19	Maximum Nozzle Plume Expansion	16-53

LIST OF ILLUSTRATIONS (Cont'd)

Figure No.	Title	Page
16-20	Nozzle Plume Temperature in the Vicinity of the Landing Gear	16-54
16-21	Window Heat Loss from a Single Glass Pane (Lunar Cold Side)	16-56
16-22	Window Heat Loss From a Double Glass Pane (Lunar Cold Side)	16-57
16-23	Electrical Heater Power Required for Window Defogging	16-58
16-24	Properties of Coating PE-81E	16-60
16-25	Effect of Outer Surface Emissivity on Maximum Window Heat Loss	16-61
16-26	Heat Transferred Through Windows as a Function of Window Properties	16-62
16-27	Effect of Vehicle Surface Emissivity on Heat Loss from Cabin Walls (Lunar Cold Side Operation)	16-64
16-28	Effect of Insulation Thickness on Heat Loss from Cabin Walls (Lunar Cold Side Operation)	16-65
16-29	Effect of Vehicle Surface Properties on Heat Transferred through the Cabin Walls (Lunar Hot Side)	16-67
16-30	Effect of Insulation Thickness on Heat Transferred through the Cabin Walls (Lunar Hot Side)	16-68
16-31	Weight Penalty to Cabin Wall for Thermal Protection (Lunar Hot Side)	16-69
16-32	Vehicle Thermal Load Summary	16-72
16-33	Heat Transport Loop Performance Summary	16-74
16-34	Radiator Control Performance with Fluid By-Pass and Regenerative Heat Exchanger	16-78
16-35	Radiator Equilibrium Temperature Variation (Lunar Hot Side)	16-79
16-36	Radiator Outlet Temperature Vs. Lunar Hot Side Stay Time	16-81

LIST OF ILLUSTRATIONS (Cont'd)

Figure No	Title	Page
16-37	Water Boiler Flow Rate Vs. Lunar Hot Side Stay Time	16-82
16-38	Optimum Space Radiator - H_2O Boiler Combination Vs. Hot Side Stay Time	16-83
16-39	Optimum Radiator Area Vs. Lunar Hot Side Stay Time	16-84
16-40	Recommended ECS Schematic	16-89
16-41	Suit Circuit Leak Check Schematic	16-97
16-42	Suit Circuit Purge Schematic	16-98
16-43	Suit Circuit Operation with an Unpressurized Cabin	16-100
16-44	Suit & Cabin Circuit Operation with Pressurized Cabin	16-102
16-45	Water Wick Boiler Performance	16-107
16-46	Freeze Point & Coolant Weight for Water-Ethelyene Glycol Coolant Solutions	16-108
16-47	Weight of Space Vehicle Water Tankage	16-112
16-48	Weight of Supercritical O_2 Storage Systems	16-114
16-49	Weight of Lithium Hydroxide & Charcoal Canister	16-115
16-50	Effect of Lunar Cold Side Stay Time on ECS Components Weight	16-119
16-51	ECS Development Schedule	16-128
16A-1	Definition of Crew Activity Index	16A-2
16A-2	Programmed Activity Cycle for 24 Hour Runs	16A-2
16A-3	Programmed Activity Cycle for Emergency	16A-3
16A-4	Schematic of Representative Atmosphere Control System	16A-4
16A-5	Analog Performance of Representative System	16A-6
16A-6	Analog Performance of Representative System (Emergency Activity)	16A-7

LIST OF ILLUSTRATIONS (Cont'd)

Figure No.	Title	Page
16A-7	Effect of Varying Weight Flow on System Performance	16A-9
16A-8	Effect of Varying Total Pressure on System Performance	16A-10
16A-9	Effect of Wall Temperature on System Performance	16A-11
16A-10	Diagram of Analog Program for the Atmosphere Control System	16A-14

LIST OF TABLES

Table No.	Title	Page
16-I	Man's Average Daily Inputs and Outputs	16-6
16-II	Recommended LEM Environmental Control Design Limits	16-7
16-III	LEM oxygen Requirements	16-14
16-IV	Comparison of Oxygen Storage Methods	16-15
16-V	Advantages and Disadvantages of CO_2 Removal Methods	16-18
16-VI	Comparison of Water Separators	16-20
16-VII	Summary of Effects and Removal Temperatures of LEM Heat Loads	16-22
16-VIII	Comparison of Methods of Heat Removal from Fluids	16-23
16-IX	Comparison of Equipment Thermal Control Methods	16-24
16-X	Weight Penalty for Returning Urine & Wash Water to Earth	16-42
16-XI	Summary of Cabin Heat Loads	16-71
16-XII	ECS Component List	16-91
16-XIII	ECS Power Requirements Summary	16-110
16-XIV	ECS Changes for Hot Side Landing	16-121
16-XV	ECS Warning Lights	16-122
16-XVI	ECS Indicators & Recorders	16-123
16-XVII	ECS Controls Summary	16-124
16-XVIII	Ground Checkout Requirements	16-125
16A-I	Analog Program Inputs and Outputs	16A-21

ENVIRONMENTAL CONTROL SYSTEM

16.0 ENVIRONMENTAL CONTROL SYSTEM

16.1 INTRODUCTION

This volume presents the results of the environmental control portion of the Lunar Excursion Module (LEM) Study Program. The objectives of the environmental control system study were two-fold.

(a) To provide the environmental control system parametric data required for evaluation of the possible vehicles and missions.

(b) To provide a preliminary design of the best practical environmental control system for the selected vehicle and mission.

In order to achieve these objectives the study program was conducted in two parts. Initially, component and system trade studies were conducted to define the best systems for the full range of vehicles and missions. This range included a crew size of 1 and 2 men, a mission duration of 1 to 7 days and a range of possible lunar landing sites (i.e., equatorial lunar noon and lunar midnight). Weights, volumes, power requirements and operation of environmental systems for vehicles and missions covering the above range were defined. These inputs were used in trade studies to establish the most desirable vehicle and mission. Once the vehicle and mission were defined, additional detailed studies were conducted to define a recommended environmental control system (ECS) and to determine solutions for special environmental problems not directly connected with internal environmental control.

Participation in the program by subcontractors of environmental systems and components was prohibited under the restrictions established by NASA. Physical data on existing and planned components were obtained from subcontractor data available from previous programs and from the results of current and past government research and development studies. In addition, the contractor responsible for the design and manufacture of Apollo, Gemini, and Mercury ECS equipment was visited to determine the development status of these systems.

16.2 SUMMARY

The LEM Environmental Control study results and supporting data are presented in this report. The technical discussion in this volume is sub-divided into the following major sections.

(a) Life Support Requirements

(b) Component & Configuration Trade Studies

(c) System Parametric Studies

(d) Analysis of Environmental Control for the Selected Vehicle and Mission

(e) Recommended Environmental Control System Description

A summary of the conclusions and material covered in each of these sections is presented in the paragraphs that follow.

Trade studies were conducted to define Life Support Requirements which would be compatible with man and still not impose a severe penalty on the ECS. These studies resulted in the selection of a 5 psia pure oxygen atmosphere with a normal carbon dioxide concentration below 2.6 mm Hg and a temperature and humidity range of 68 to 76°F and 30 to 60 percent respectively. These limits will satisfy man's requirements and can be maintained with a lightweight and reliable environmental control system.

Component and configuration trade studies were conducted early in the program to determine the most suitable methods for satisfying the above requirements. Based on these trade studies, the following components and configurations are recommended for the LEM.

(a) Oxygen Storage - Supercritical cryogenic storage is recommended for the normal oxygen supply and high pressure gaseous storage is recommended for the emergency supply. However, the significant weight savings which can be obtained by using two phase oxygen storage with thermal expulsion makes this method worthy of additional consideration.

(b) Carbon Dioxide (CO_2) Removal - The recommended method for CO_2 removal is use of lithium hydroxide canisters. This is a proven concept and provides the lightest, most reliable, and most flexible method for meeting the LEM requirements.

(c) Humidity Control - A system which provides for condensation of the water in the temperature control heat exchanger with subsequent separation of the water from the airstream by a centrifugal separator is recommended for humidity control.

(d) Contaminate Control - Activated charcoal and a Hopcalite catalytic burner provide the most practical methods for removal of odors and

trace contaminates. For a one-day mission the catalytic burner will not be required; however, for growth to a 7-day mission capability, the burner may be required. Further evaluation of the trace gas build-up problem is needed to definitely establish a requirement for the catalytic burner.

(e) Internal Thermal Control - A water-ethylene glycol heat transportation loop and space radiator system was selected as the method for internal thermal control. The electrical equipment is cooled by glycol cold plates and the cabin atmosphere is cooled by two air-to-glycol heat exchangers. A water boiler system is recommended for radiator augmentation during peak heat load conditions and during operation on the lunar hot side.

(f) Thermal Control Outside Pressure Shell - In most cases, temperature limited equipment, lines, and structure outside the pressure shell can be controlled within acceptable limits by passive means (i.e., thermal control coatings and super insulation). Further study on each individual item is required to define specific details.

(g) Air Lock - Inclusion of an air lock provides the lightest means of ingress and egress from the space cabin if more than 4 trips outside the LEM are contemplated Therefore, an air lock is not required for a one-day lunar stay time (2 to 4 trips outside vehicle). However, in order to provide growth capability beyond one day, an air lock has been included in the LEM.

(h) Translunar Operation - Studies show that during the translunar phase of the mission the environmental control system should be inactive and that the entire vehicle should be conditioned by the Command Module ECS. This allows the crew to utilize the LEM volume during the translunar phase at a minimum penalty to the over-all mission.

(i) Fuel Cell Integration - Trade studies show that over-all advantages are obtained by cooling the fuel cell with the ECS thermal loop and by utilizing the water produced in the fuel cell for cooling and drinking. A preliminary comparison showed that no advantage is obtained by integrating the fuel cell and ECS cryogenic supplies.

(j) Waste Handling - Significant weight savings are possible if waste material is off-loaded on the lunar surface prior to rendezvous with the Command Module. Further study is needed to define sanitation procedures which will make this technique acceptable.

Using the results of the above component trade studies, parametric studies of various environmental control systems were conducted to provide ECS data for mission and vehicle selection. Preliminary systems for 1 and 2 man crews, 1 to 7 day lunar stay times, lunar hot side operation and lunar cold side operation were defined. Weights, volumes and power requirements of these systems were compiled and supplied as inputs to the over-all vehicle synthesis. These studies indicated that the weight penalty for providing growth to a 7-day mission on the cold side of the moon is not excessive but that the penalty for lunar hot side operation is significant. Based on these results (and results from other LEM system studies) the

recommended 24 hour cold side vehicle with growth capability to 7 days on the cold side and 2 days on the hot side was derived.

Following definition of the vehicle and mission, more detailed studies were conducted to evaluate specific ECS problems. These included cabin leakage, exhaust nozzle plume heating, structural heat leakage, window heat leakage, detailed internal thermal control loop analysis and an analog study of the atmosphere control system.

Based on the results of all the above trade studies and detailed analyses, the recommended configuration for the environmental control system was defined. A detailed schematic, weight and weight justification, operational modes, changes for growth missions, controls and displays and ground checkout requirements are presented. It is concluded that the recommended system will satisfy the LEM requirements and can be designed, constructed and demonstrated within the limits of the LEM master schedule. The ECS does not represent a major problem area to the LEM development.

LIFE SUPPORT REQUIREMENTS

16.3 DISCUSSION

16.3.1 Life Support Requirements

The environmental requirements established for the LEM are summarized in Tables 16-I and 16-II. Table 16-I presents man's average daily inputs and outputs for both shirtsleeve and pressure suit environments and Table 16-II presents the recommended limits for atmosphere control of the LEM. These requirements have been used for the vehicle and subsystem design. In defining the requirements shown by Table 16-II, the following factors were considered:

(a) Environmental stress data (available in research literature).

(b) Established limits for Apollo, Mercury, Gemini, etc.

(c) ECS Weight, complexity, reliability, and development status.

(d) The margin of safety between the recommended zone and the tolerable limits. Wherever possible a wide margin was provided.

(e) The severity of the consequences of extending the tolerable limits. These data provided a guideline for establishment of the width or scope of the safety margin.

Subsequent to releasing the limits shown by Table 16-II, trade studies were conducted on each parameter to determine the advantages and disadvantages of the recommended zone. These trade studies are summarized in the following paragraphs.

16.3.1.1 Cabin Atmosphere

A 98 - 100% oxygen (single gas) system with total pressures of 5.0 psia, normal limits, and 3.5 psia, emergency limits is recommended. Figure 16-1 depicts this oxygen-pressure zone in comparison with the Apollo oxygen-pressure zone. Note that both are well within the margins of safety and within the unimpaired performance envelope. The advantages of the 5.0 psia pure oxygen system when compared with a higher pressure mixed gas system are:

(a) Reduces the explosive decompression hazard in the event of a cabin rupture.

(b) Decreases leak rate which conserves cabin pressurization gases.

(c) Saves ECS weight and increases reliability by eliminating the need for nitrogen storage and complicated partial pressure controls.

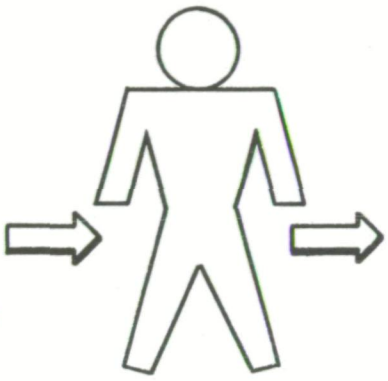
(d) Reduces probability of bends in the event of a rapid cabin decompression.

TABLE 16 – IA

MAN'S AVERAGE DAILY INPUTS & OUTPUTS (SHIRT SLEEVE ENVIRONMENT)

AVERAGE INPUTS

ITEM	WT (LBS)
FOOD	1.6
WATER – IN FOOD	1.9
IN DRINKS	2.9
OXYGEN	2.0
TOTAL	8.4 LBS



AVERAGE OUTPUTS

ITEM	WT (LBS)
FECAL SOLID	.3
VAPOR – SWEAT	1.5
– BREATH	.8
LIQUID – URINE	2.9
– FECAL	.4
CARBON DIOXIDE	2.5
TOTAL	8.4 LBS

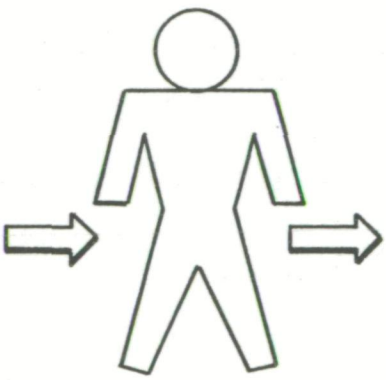
(.8 LBS/DAY H_2O FORMED IN FOOD METABOLISM)

TABLE 16 – IB

MAN'S AVERAGE DAILY INPUTS AND OUTPUTS (PRESSURE SUIT CONDITIONS)

AVERAGE INPUTS

ITEM	WT (LBS)
FOOD	1.6
WATER – IN FOOD	2.2
IN DRINKS	7.7
OXYGEN	2.0
TOTAL	13.5 LBS



AVERAGE OUTPUTS

ITEM	WT (LBS)
FECAL SOLID	.3
VAPOR – SWEAT	6.5
BREATH	.9
LIQUID – URINE	2.9
FECAL	.4
CARBON DIOXIDE	2.5
TOTAL	13.5

(.8 LBS/DAY H_2O FORMED IN FOOD METABOLISM)

TABLE 16-II RECOMMENDED LEM ENVIRONMENTAL CONTROL DESIGN LIMITS

ENVIRONMENTAL FACTOR	NORMAL LIMITS	NOMINAL MAXIMUM LIMITS	EMERGENCY LIMITS
TOTAL PRESSURE	5.0 psia 258 mm Hg		3.1 psia (Minimum) (160 mm Hg)
OXYGEN PARTIAL PRESSURE PERCENT – OXYGEN	255–258 mm Hg 99 – 100%	253–258 mm Hg 98 – 100%	160 mm Hg (Minimum) 95 – 100%
CO_2 PARTIAL PRESSURE PERCENT CO_2	0–2.6 mm Hg 0%	0–5.2 mm Hg 0–2%	13 mm Hg (Maximum) at 5 psia Total Pressure 5% at 5 psia Total Pressure
TEMPERATURE RELATIVE HUMIDITY	68° F – 76°F at 30 – 60% R.H.	SEE TABLE BELOW FOR SHORT TERM TOLERANCES	

NOMINAL AND EMERGENCY TEMPERATURE LIMITS FOR SHORT PERIODS OF EXPOSURE (30–60% RELATIVE HUMIDITY)

EXPOSURE TIME	RH	NOMINAL MAXIMUM LIMITS	EMERGENCY LIMITS
90 MIN.	30	64° – 98°	58° – 112°
	60	63° – 91°	58° – 100°
30 MIN.	30	60° – 102°	44° – 122°
	60	60° – 98°	44° – 109°
15 MIN.	30	55° – 115°	33° – 127°
	60	55° – 100°	33° – 112°

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

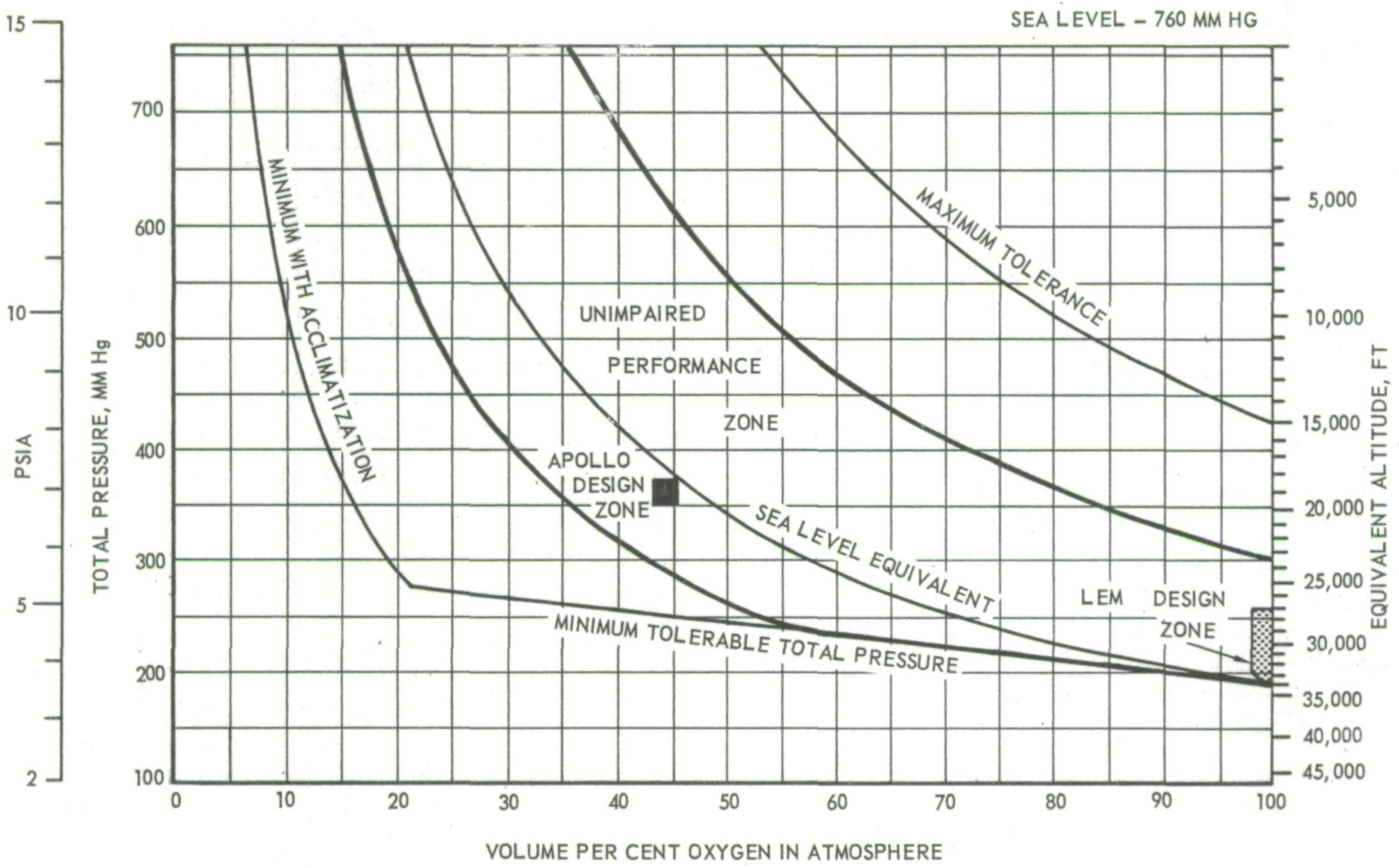


Figure 16–1 OXYGEN – PRESSURE EFFECTS

The disadvantages of the 5.0 psia pure oxygen system when compared with a higher pressure mixed gas system are as follows:

(a) Slightly higher fire hazard if meteoroid penetration occurs.

(b) Somewhat more severe nasal passage dryness and irritation.

The advantages of the recommended 5 psia pure oxygen system far outweigh the disadvantages. (The 5 psia pure oxygen system has a precedent in Mercury, Gemini, and in the prototype Apollo extra vehicular space suit.) No physiological problem due to use of this system is currently foreseen.

16.3.1.2 Carbon Dioxide (CO_2) Concentration

The recommended upper limits for CO_2 concentration in a 5 psia atmosphere are 2.6 mm Hg normal, 5 mm Hg nominal maximum and 13 mm Hg emergency. These concentrations correspond to 1%, 2%, and 5% respectively under the 5 psi total pressure conditions. These limits are essentially consistent with the approved Apollo limits.

The emergency limits are recommended for extremely short periods because CO_2 exchange in the lung alveoli ceases at about 12 to 15 mm Hg at 5 psia and acidosis will follow. Manhigh II experiments indicate that 14 mm Hg CO_2 for approximately 7 minutes resulted in reduced efficiency (reading errors) to the degree that ground monitors instituted emergency measures to reduce the CO_2 concentration.

The LEM crew may not contribute as much CO_2 output under null gravity as under 1 G or higher accelerations. Consequently, the CO_2 content in the cabin is expected to be normally somewhat lower than in the earth's atmosphere.

16.3.1.3 Temperature-Relative Humidity

A temperature range of 68^{o} F to 76^{o} F and a relative humidity range of 30% to 60% is recommended for the LEM.

The selection of these limits was based on a determination of the unimpaired performance envelope for dry bulb temperatures at various relative humidities. Then, within this envelope, a zone was recommended which is most easily achieved from an engineering standpoint and which results in a range of relative humidities sufficient to reduce the undesirable dry oxygen nasal irritation effects.

Figure 16-2 depicts the recommended humidity-temperature design zone for a pure oxygen, 5 psia system. Also, depicted is the Lovelace unimpaired performance envelope. Note that the recommended design zone falls entirely within the unimpaired performance envelope.

Figure 16-3 shows the slightly higher temperature-humidity zone recommended by NASA for Apollo which is superimposed on an O_2-N_2

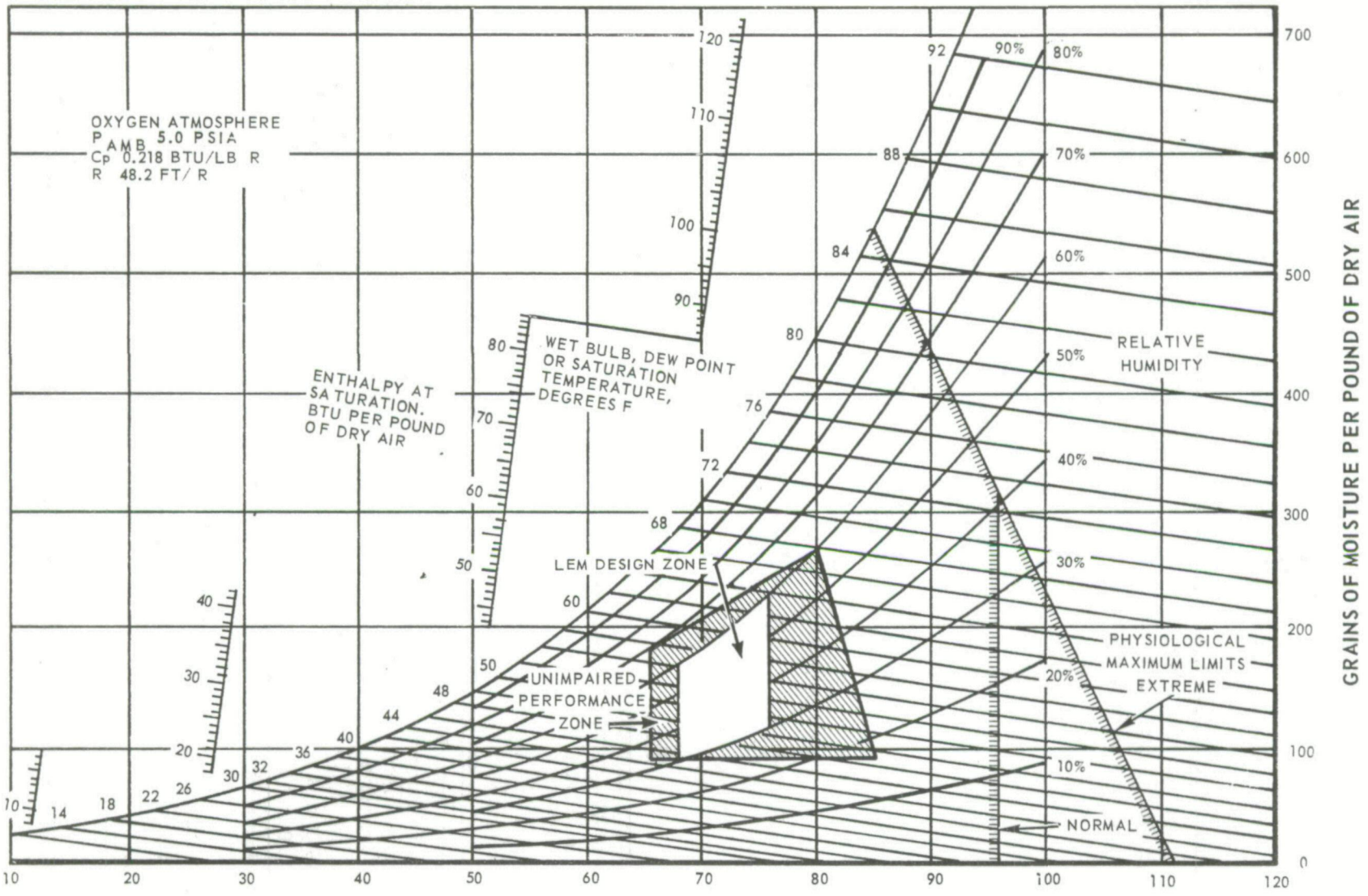


Figure 16–2 PSYCHROMETRIC CHART-OXYGEN ATMOSPHERE, 5.0 PSIA (27,000'PRESSURE ALTITUDE) TOTAL PRESSURE

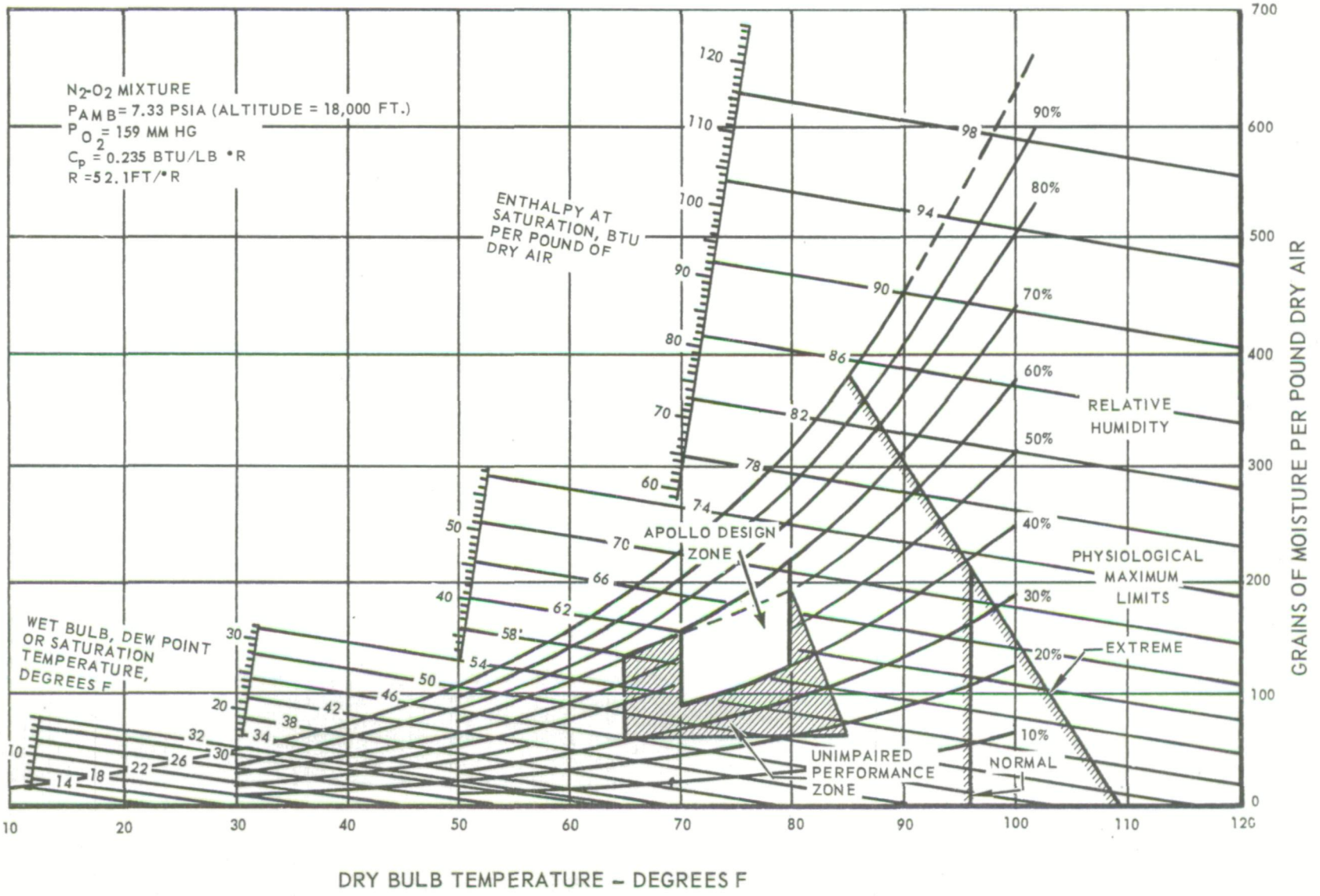


Figure 16–3 PSYCHROMETRIC CHART—OXYGEN – NITROGEN MIXTURE, 7.33 PSIA (18,000' PRESSURE ALTITUDE) TOTAL CABIN PRESSURE; 3.1 PSIA (SEA LEVEL PRESSURE ALTITUDE) PARTIAL PRESSURE OXYGEN

mixed gas chart at 7.3 psia. Note that the Apollo design zone falls slightly beyond the unimpaired performance zone in the higher temperature-relative humidity range. The LEM system is capable of maintaining the relative humidity below these values and well within the unimpaired performance envelope.

COMPONENT AND CONFIGURATION TRADE STUDIES

16.3.2 Component and Configuration Trade Studies

Comparative analyses of the various methods of accomplishing environmental control were conducted in order to define the most suitable components for the full range of LEM requirements. The criteria used for comparing these components are listed below in the order of importance assigned:

(a) Performance

(b) Reliability

(c) Development Status

(d) Weight

(e) Growth capability

(f) Size

(g) Cost

A discussion of the various components and the results of the comparative analyses are presented in the following paragraphs.

16.3.2.1 Oxygen Storage

An oxygen storage system is required to provide the metabolic inputs and to replace cabin gas lost through leakage, cabin depressurization, and air lock operations. A summary of the weights of oxygen required for both 24-hour and 7-day missions is shown by Table 16-III. From this table it can be seen that 49.5 lbs and 126 lbs of oxygen are required in the primary oxygen storage container for the 24-hour and 7-day missions respectively.

The five methods considered for storage of the above oxygen requirements are as follows:

(a) Gaseous storage

(b) Supercritical cryogenic storage

(c) Subcritical cryogenic-positive expulsion (i.e., bladder)

(d) Subcritical cryogenic-thermal expulsion (i.e., fluid withdrawn as mixed gas and vapor with heat input assuring delivery of gas to the cabin).

(e) Superoxide chemicals

Table 16-IV presents a comparison of these methods based on the selected evaluation criteria. Data for this comparison were obtained from References 1, 2 and 3. Figure 16-4 presents a weight comparison of the different methods. From Figure 16-4 and Table 16-IV it is seen that the lightest method for the primary storage system (50 to 126 lbs of O_2) is sub-

TABLE 16-III LEM OXYGEN REQUIREMENTS

ITEM	WEIGHT POUNDS	
	24 HR. MISSION	7 DAY MISSION
1. METABOLIC REQUIREMENTS 2 LBS/MAN DAY	4	28
2. LEAKAGE – .1 LBS/HR.	2.4	16.8
3. CABIN REPRESSURIZATIONS – 2 AT 400 FT^3 PER REPRESSURIZATION	22.4	22.4
4. AIRLOCK OPERATIONS – 38 FT^3 AIRLOCK (1.05 LBS. PER OPERATION)	4.2	16.8
5. 50% EXCESS OF ABOVE	16.5	42
TOTAL	49.5	126

TABLE 16 – IV COMPARISON OF OXYGEN STORAGE METHODS

ITEM AND LEGEND	7,500 psia GASEOUS STORAGE		SUPERCRITICAL		SUBCRITICAL – BLADDER		SUBCRITICAL – THERMAL		SUPEROXIDE (KO_2)	
	50 LBS.	126 LBS.	50 LBS.	126 LBS.	50 LBS.	126 LBS.	50 LBS.	126 LBS.	50 LBS.	126 LBS.
STORAGE EQUIPMENT WEIGHT – LBS./LBO_2	3.5	3.5	.56	.52	.43	.28	.09	.08	3.12	3.12
SPECIFIC VOLUME – FT^3/LBO_2	.035	.035	.018	.017	.018	.017	.018	.017	.076	.076
HEAT ABSORBED BTU/LB.	NEGLIGIBLE		135		178		178		–2560	
RELIABILITY (RELATIVE)	EXCELLENT		EXCELLENT		GOOD		GOOD		GOOD	
DEVELOPMENT STATUS 5 – DEVELOPED 1 – HIGH RISK	5		4		2		2		3	
MISSION GROWTH POTENTIAL (RELATIVE)	POOR		EXCELLENT		EXCELLENT		EXCELLENT		FAIR	
OTHER	-----		-----		-----		-----		ABSORBS CO_2 AND WATER	

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

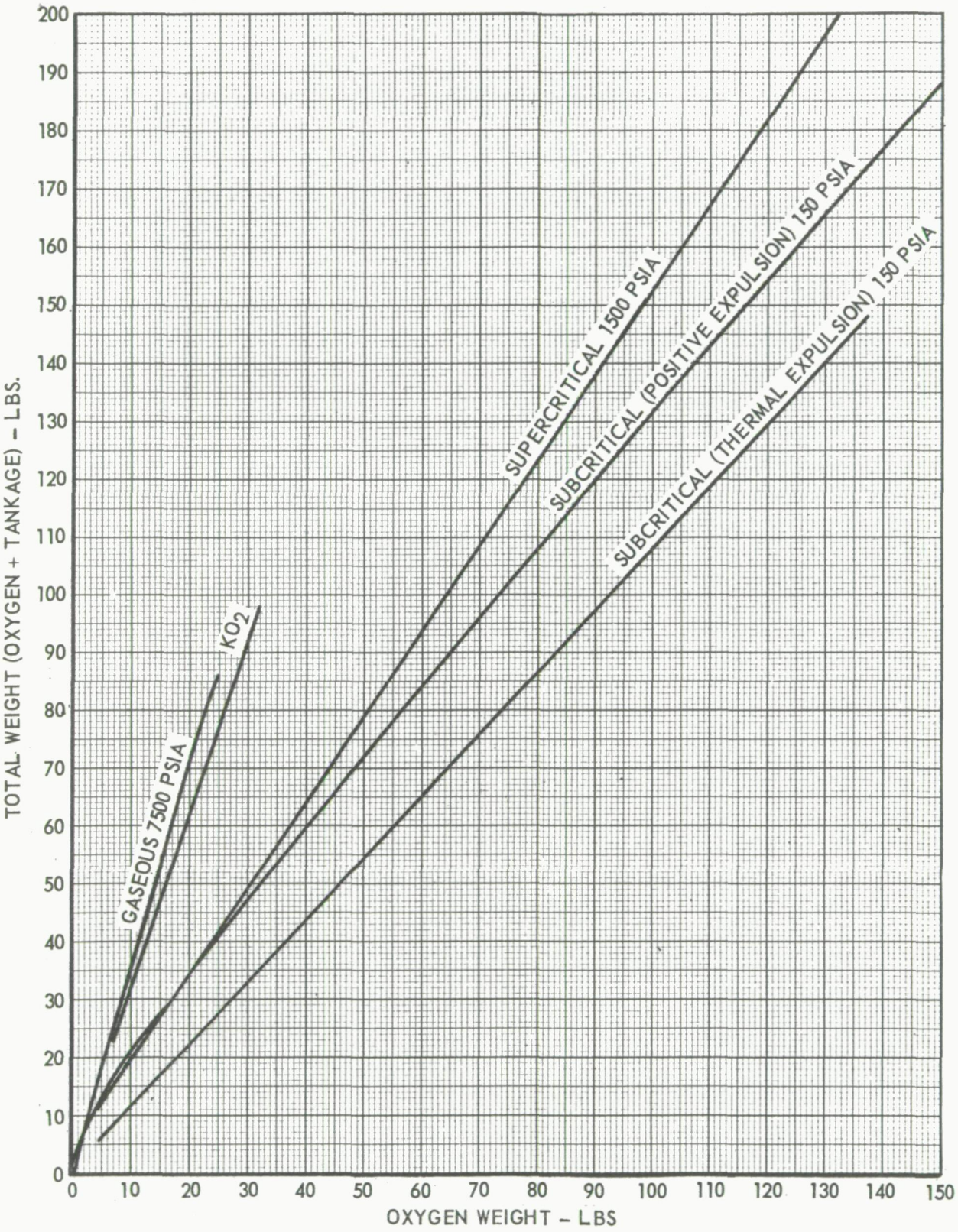


Figure 16-4 COMPARISON OF OXYGEN STORAGE SYSTEM WEIGHTS

critical two-phase storage with thermal inputs for expulsion. This method may very well be adopted if further development is conducted to prove the method under space conditions. At the present time, however, only limited experience is available with this type system.

Bladder expulsion provides the next lightest storage method, however, reliable multi-cycle bladder systems have not been developed. Gaseous and potassium superoxide storage systems are very heavy for the oxygen quantities required. Considering all criteria, the supercritical storage technique is recommended for the primary oxygen storage system at this time. High pressure gaseous storage is recommended for the emergency supply (approximately 4 lbs.). Prior to selection for actual construction, the development status of the two-phase thermal expulsion method should be re-accessed, since all indications are that this will be the ultimate lightweight system for oxygen storage in the future.

16.3.2.2 Carbon Dioxide (CO_2) Removal

Numerous methods have been evaluated by industry for control of the CO_2 concentration in manned spacecraft. The methods considered applicable to the LEM are non-regenerative chemical absorption (LiOH), regenerative chemical absorption, molecular sieves, freeze out and membrane diffusion. Table 16-V lists the major advantages and disadvantages of these CO_2 removal methods.

Considering all of the methods compared in Table 16-V, the logical choice for the LEM is the proven concept of non-regenerative chemical absorption (LiOH). The estimated total weight of an LiOH system for the LEM is 16 lbs for 2 man-days and 80 lbs for 14 man-days. This method is the lightest, most reliable and has growth capability for increased mission time simply by the addition of more canisters to the initial vehicle stores. Another advantage which improves the weight picture even more is that "used" canisters can be left on the lunar surface thereby reducing the lunar launch propulsion weight by approximately 1.5 lbs per lb of used canister discarded.

The use of Lithium Oxide (LiO_2) to improve the LiOH technique is now being investigated for the Apollo Program. With this method, water in the air stream reacts with the LiO_2 to form LiOH which in turn removes the CO_2 from the air stream. This approach appears to offer a weight saving over the LiOH system and should be given further consideration when results from the current investigation are available. At the present time, the comparison of methods shows that the best overall approach is use of LiOH canisters.

16.3.2.3 Humidity Control

Possible methods for removal of water vapor from the cabin atmosphere include condensation and collection, chemical absorption, and molecular sieves (adsorption). An important consideration in the choice of the system to be used is whether the water may be conveniently collected for re-use. This is especially true for lunar orbit and lunar hot side missions because the collected water may be used to augment the radiator in order to obtain the desired cooling. The latter two methods described above would

TABLE 16 – V ADVANTAGES AND DISADVANTAGES OF CO_2 REMOVAL METHODS

	NON-REGENERATIVE CHEMICAL ABSORPTION (LiOH)	REGENERATIVE CHEMICAL ABSORPTION (Ag O)	MOLECULAR SIEVES (MS)	FREEZE OUT	MEMBRANE DIFFUSION
ADVANTAGES	1. SIMPLE SYSTEM 2. AVAILABILITY – SYSTEM OPERATING ON MERCURY. UNDER DEVELOPMENT FOR APOLLO. 3. FLEXIBILITY – MISSION CAPABILITY EXTENDED BY ADDING CANISTERS. 4. NO POWER REQUIRED EXCEPT CIRCULATING FAN. 5. EXCELLENT STORABILITY.	1. FIXED WEIGHT REGARDLESS OF MISSION TIME. 2. GOOD STORAGE CHARACTERISTICS. 3. CO_2 SEPARATED. MAY BE REUSED IF DESIRED.	1. FIXED WT. REGARDLESS OF MISSION TIME. 2. CO_2 IS SEPARATED AND MAY BE RE-USED IF DESIRED. 3. COMPACT.	1. FIXED WEIGHT REGARDLESS OF MISSION TIME. 2. CO_2 SEPARATED. MAY BE RE-USED IF DESIRED.	1. RELATIVELY SIMPLE SYSTEM. 2. FIXED WEIGHT REGARDLESS OF MISSION TIME. 3. CO_2 SEPARATED. MAY BE RE-USED IF DESIRED.
DISADVANTAGES	1. DUSTING PROBLEM. 2. WEIGHT INCREASES WITH MISSION TIME. 3. RELATIVELY HIGH VOLUME PENALTY.	1. REQUIRES EXCESSIVE POWER. 2. SMALL TEMPERATURE DIFFERENCE BETWEEN REGENERATIVE TEMP. AND DECOMPOSITION TEMP. CLOSE CONTROL REQUIRED. 3. COMPLICATED SYSTEM. LOW RELIABILITY.	1. AFFINITY OF MS TO WATER VAPOR. AIR MUST BE DRIED PRIOR TO CO_2 REMOVAL BY MS BED. 2. COMPLICATED SYSTEM. LOW RELIABILITY	1. REQUIRES LOW TEMPERATURE HEAT SINK. 2. COMPLICATED SYSTEM. LOW RELIABILITY. 3. EXTENSIVE DEVELOPMENT REQUIRED.	1. LARGE DIFFUSION AREA REQUIRED CAUSING LARGE VOLUME PENALTY. 2. EXTENSIVE DEVELOPMENT REQUIRED.

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

require both power and complicated supporting systems for collection of the water. The lightest and most reliable method for humidity control is by condensation of the water vapor in the temperature control heat exchangers. The water condensed must then be collected in a water separator.

A variety of water separation techniques have been developed and tested by equipment manufacturers. These include the sponge type, the air driven centrifugal type, the elbow type and the cyclone type (Reference 2). Table 16-VI presents a discussion and comparison of these water separators. Recent improvements in air driven centrifugal separators makes their use very attractive except for their susceptability to solid particles. The sponge type separator appears undesirable due to the complexity of the squeezing mechanism and the loss of the air trapped in the sponge during squeezing.

The air driven centrifugal separator has been selected as the best over-all solution. This selection should be reviewed prior to actual construction to evaluate possible new developments.

16.3.2.4 Contaminate Control

Body waste, leakage of fluid stores and chemical reactions can cause excessive build up of trace contaminates in long duration sealed space cabins. The magnitude of this problem is relatively undefined at the present time. Build up of trace contaminates during manned sealed cabin operation on earth has been studied (Reference 4), however, the results appear to be applicable only to the particular system used. Extrapolation of these results to an actual space cabin would yield inconclusive data. Mercury experience has shown that simple filtration with activated charcoal is sufficient for missions of approximately one day. Activated charcoal has the characteristic of selectively absorbing many of the organic gases in amounts up to its own weight. According to Reference 5, 1/4 of an ounce of charcoal per man-hour is sufficient. Other possible solutions to the odor control problem include electrostatic filters and superoxides. Both these methods are considerably heavier than the simple, proven activated charcoal approach. Therefore, use of activated charcoal is recommended for the LEM.

For a growth mission of 7 days the activated charcoal alone may not be sufficient. Certain gases such as methane are not absorbed by activated charcoal. In order to provide a measure of protection against build-up of gases not absorbed by charcoal, a Hoplicate catalytic burner has been included in the recommended system. This burner would be operated at cabin temperature until the presence of contaminates is indicated by smell or by the gas analyzer. When this occurred, the catalytic burner would be activated and would operate at approximately 700°F as long as the contaminate trace was detectable. This temperature is sufficient to remove expected trace gases.

16.3.2.5 Internal Thermal Control

Inclusion of a method of effective thermal control of the LEM is necessitated by the relatively low thermal tolerance of both crew members and on-board equipment. Thermal control requires: (1) removal of heat from the local source at the required temperature level, (2) transportation of the

TABLE 16 – VI COMPARISON OF WATER SEPARATORS

ITEM	SPONGE	AIR DRIVEN CENTRIFUGAL	ELBOW	CYCLONE
PRINCIPLE OF OPERATION	WATER DROPLETS IN THE AIR STREAM ARE DEPOSITED ON THE SPONGE AND RETAINED BY CAPILLARY ACTION. PERIODIC SQUEEZING OF THE SPONGE IS REQUIRED.	AIR & WATER DROPLETS STRIKE A TURBINE WHEEL CAUSING ROTATION. DUE TO INERTIA, THE WATER COLLECTS ON THE OUTSIDE WALL AND IS REMOVED.	AIR & WATER DROPLETS ARE MADE TO TURN THROUGH A LARGE ANGLE. THE INERTIA OF THE WATER CAUSES IT TO COLLECT ON THE OUTSIDE WALL OF THE BEND. PASSAGES ARE PROVIDED ON THE OUTSIDE WALL FOR WATER REMOVAL.	AIR & WATER AT A RELATIVELY HIGH SPEED ARE INTRODUCED INTO THE SEPARATOR. STATIC VANES INDUCE FREE VORTEX FLOW INSIDE THE SEPARATOR. THE WATER IS COLLECTED ON THE SIDES AND IS WITHDRAWN AT THE DOWNSTREAM END OF THE SEPARATOR.
ADVANTAGES	● DEVELOPED-SYSTEM ON PROJECT MERCURY. ● NOT SIGNIFICANTLY AFFECTED BY SOLID PARTICLES. ● POSITIVE H_2O EXPULSION. ● GOOD EFFICIENCY.	● NO POWER REQUIRED. ● NO LOSS OF AIR IN REMOVING H_2O. ● LIGHTWEIGHT. ● HIGH EFFICIENCY.	● NO POWER REQUIRED. ● LIGHTWEIGHT ● NOT AFFECTED BY SOLID PARTICLES.	● NOT AFFECTED BY SOLID PARTICLES.
DISADVANTAGES	● LOSS OF AIR WHEN SPONGE IS SQUEEZED TO EXPELL H_2O. ● REQUIRES POWER AND SQUEEZE MECHANISM.	● SOLID PARTICLES MAY CAUSE BINDING. ● LOW Δp FOR H_2O WITHDRAWAL.	● LOW EFFICIENCY OF SEPARATION. ● LOW Δp FOR H_2O WITHDRAWAL.	● HIGH POWER REQUIRED TO OVERCOME PRESSURE DROP OF AIR. ● LOW Δp FOR H_2O WITHDRAWAL.

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

heat away from the source, and (3) dissipation of the heat from the spacecraft. These problems are examined separately in the following paragraphs.

16.3.2.5.1 Heat Loads and Removal Methods

Heat sources and/or sinks comprising the thermal load within the LEM can be categorized into the following five classes:

(a) Crew metabolic

(b) Structural leakage

(c) Electrical power consuming equipment

(d) Electrical power generation system (fuel cells)

(e) Cryogenic fluids.

The effects of these heat loads and the required temperature levels for heat addition or removal are summarized in Table 16-VII. From this table it can be seen that thermal control of all of the above loads requires either a change in fluid temperature level or a change in equipment temperature. These requirements are considered separately in the following paragraphs.

Possible methods of controlling fluid temperatures are listed and compared in Table 16-VIII. The length of the anticipated missions (1 to 7 days)dictates a closed or semi-closed thermal control loop, precluding the possiblity of simply venting hot gases overboard as the primary method of thermal control. A brief inspection of the wide variation in heat loads and external environments, as well as the degree of control required, eliminates the candidacy of passive cooling using the vehicle structure to provide the necessary thermal gradients. The pumping power penalty associated with space radiators using gaseous fluids is prohibitive for low temperature radiators and extended missions. As shown by Table 16-VIII, lightweight, compact, extended-surface heat exchangers within a fluid loop offer the most economical method of heat removal and provide close temperature control throughout a wide range of heat loads.

The high watt densities associated with rack-mounted electronic equipment necessitates a high capacity heat sink within a very restricted volume. The heat transfer methods for thermal control of the electronic equipment can be many and varied. Table 16-IX summarizes typical electronic equipment cooling methods. For each method, the number of watts per cubic inch of volume which can be supported is shown. Evaporative cooling is an excellent method when employed as a backup or short time duration scheme, but is costly in weight for mission durations of more than a few hours. Equipment conditioning by conduction and radiation using the vehicle skin as a passive radiator is unsuitable because of the extreme variation in external environment. The cold-plate cooling method offers compactness, light weight, full system capability, and good control throughout the range of anticipated equipment heat loads, and, in addition, lends itself well to the rack-mounting technique. Cold-plates may be designed for use

TABLE 16 - VII
SUMMARY OF EFFECTS AND REMOVAL TEMPERATURES
OF LEM HEAT LOADS

HEAT LOAD	PRIMARY EFFECT OF LOAD	REQUIRED TEMPERATURE LEVEL OF HEAT REMOVAL OR ADDITION
1. Crew Metabolic Latent Sensible	Increases cabin and/or suit relative humidity Increases temperature of cabin atmosphere	 40-50°F 40-76°F
2. Structural Leakage	Increases or decreases cabin wall temperature causing increased or decreased cabin gas temperatures.	40-76°F
3. Electrical Power Consuming Equipment	Increases temperature of equipment. This adds heat to the cabin gas and to the equipment mounts.	40-76°F (cabin gas) 0-160°F (equipment and equipment mounts).
4. Fuel Cells	The fuel cells produce waste heat, water and electrical power. The waste heat and water are removed from the fuel cell by hydrogen gas. The water must be removed and the hydrogen gas must be cooled and returned to the cell.	141-500°F
5. Cryogenic Fluids	The fuel cell and ECS cryogenic storage tanks require heat addition to maintain supercritical operating pressures	-410-60°F

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

TABLE 16 – VIII COMPARISON OF METHODS OF HEAT REMOVAL FROM FLUIDS

FLUID	THERMAL CONTROL METHOD	FIXED WEIGHT PENALTY	EXPENDABLE WEIGHT PENALTY	SYSTEM CAPABILITY	RELIABILITY	DEGREE OF CONTROL	REMARKS
CABIN OXYGEN	PASSIVE TO STRUCTURE	NONE	NONE	POOR	POOR	POOR	DEPENDENT ON EXTERNAL ENVIRONMENT & ORIENTATION
	VENTING OVERBOARD	NONE	LARGE	GOOD	GOOD	GOOD	PROHIBITIVE QUANTITIVES OF EXPENDABLES REQUIRED
	HEAT EXCHANGERS AND FLUID LOOP	SMALL	NONE	GOOD	GOOD	EXCELLENT	RELATIVELY INDEPENDENT OF EXTERNAL ENVIRONMENT
	SPACE RADIATOR	SUBSTANTIAL	NONE	FAIR	GOOD	FAIR	H_2O FREEZING PROBLEMS ON COLD SIDE
CRYOGENIC STORES (O_2 & H_2)	HEAT EXCHANGERS	SMALL	NONE	GOOD	GOOD	GOOD	GOOD CONTROL AND INDEPENDENT OF ENVIRONMENT
	PASSIVE HEATING FROM ENVIRONMENT	NONE	NONE	FAIR	POOR	POOR	OVERLY DEPENDENT ON EXTERNAL ENVIRONMENT
	ELECTRICAL HEATERS	SMALL	SUBSTANTIAL		FAIR	FAIR	SIGNIFICANT POWER PENALTY ON APU
FUEL CELL HYDROGEN	HEAT EXCHANGER	SMALL	NONE	GOOD	GOOD	EXCELLENT	GOOD CONTROL FOR H_2O REMOVAL
	PASSIVE	NONE	NONE	POOR	POOR	POOR	OVERHEATING OF STRUCTURE AND COMPONENTS
	VENTING	NONE	LARGE	GOOD	GOOD	GOOD	NOT POSSIBLE TO CONTROL F.C. HYDROGEN FLOW
	RADIATOR	SUBSTANTIAL	NONE	FAIR	FAIR	FAIR	FREEZING PROBLEMS ON COLD SIDE

TABLE 16-IX COMPARISON OF EQUIPMENT THERMAL CONTROL METHODS

METHOD	CAPABILITY*	TRANSPORT METHODS	FIXED WEIGHT	EXPENDABLE WEIGHT	RELIABILITY	REMARKS
PASSIVE TO STRUCTURE	LOW – DEPENDING ON EXTERNAL ENVIRONMENT	METALLIC CONDUCTION TO VEHICLE SKIN	NONE	NONE	POOR DUE TO VEHICLE ORIENTATION REQUIREMENTS	EQUIPMENT MOUNTING & VEHICLE ORIENTATION PROBLEMS
EVAPORATIVE FLUIDS	VERY HIGH 20-23 WATTS/IN^3	FLUID ACTS AS SINK	MODERATE	LARGE	GOOD	EXCESSIVE EXPENDABLES REQUIRED FOR EXTENDED MISSIONS
DIRECT FLUIDS	HIGH 10-12 WATTS/IN^3	FLUID ACTS AS SINK	CONSIDERABLE	NONE	GOOD	FEASIBLE ONLY FOR SHORT DUTY CYCLE EQUIPMENT
INDIRECT FLUID (COLD PLATES WITHIN A FLUID LOOP)	MEDIUM 7-10 WATTS/IN^3	FLOWING FLUID	MODERATE	NONE	GOOD	GOOD EXTENDED MISSION CAPABILITY. NO EXPENDABLE WEIGHT IF FLUID IS RE-CIRCULATED
PLASTIC EMBEDMENT	LOW 0.25 WATT/IN^3	PLASTIC ACTS AS SINK	CONSIDERABLE	NONE	GOOD	FEASIBLE ONLY FOR SHORT DUTY CYCLE EQUIPMENT

*REFERENCES 6 & 7

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

with any fluid and may include change-of-phase heat sink material for emergency use should fluid flow cease.

In view of the above, extended surface and cold-plate heat exchangers are recommended as the methods for heat removal from the local source for the LEM.

16.3.2.5.2 Heat Transportation

From a study of the various required temperature levels of local heat removal, it becomes apparent that a single working fluid could serve several heat sources if the fluid were routed through sources of successively higher temperatures. This results in a substantial reduction in the total quantity of fluid required. In addition, experience has shown that when thermal control of a number of components in reasonably close proximity is required, a centralized sink system results in equipment weight and volume savings over multiple isolated systems. Therefore, an intermediate fluid to transport the heat from the heat exchangers to the vehicle heat dissipation system interface is recommended for the LEM.

Selection of the heat transport fluid requires an evaluation of candidate fluids based on the following criteria:

(a) Pumping power

(b) Specific heat

(c) Vapor pressure

(d) Freezing point

(e) Density

(f) Viscosity

(g) Material compatibility

(h) Toxicity

Figures 16-5, 16-6 and 16-7 present properties of heat transport fluids (specific heat, viscosity and density). In all cases, water or water solutions have desirable properties. The pump power requirements for heat transport fluids can be compared on the basis of the parameter $\mu/C_p^2 \rho^2$ for laminar flow and $\mu \cdot 2/C_p^{2.8} \rho^2$ for turbulent flow where:

μ = Viscosity

C_p = Specific Heat

ρ = Density

These parameters are compared for various fluids in Figures 16-8 and 16-9. From these data it is noted that pumping power for water solutions

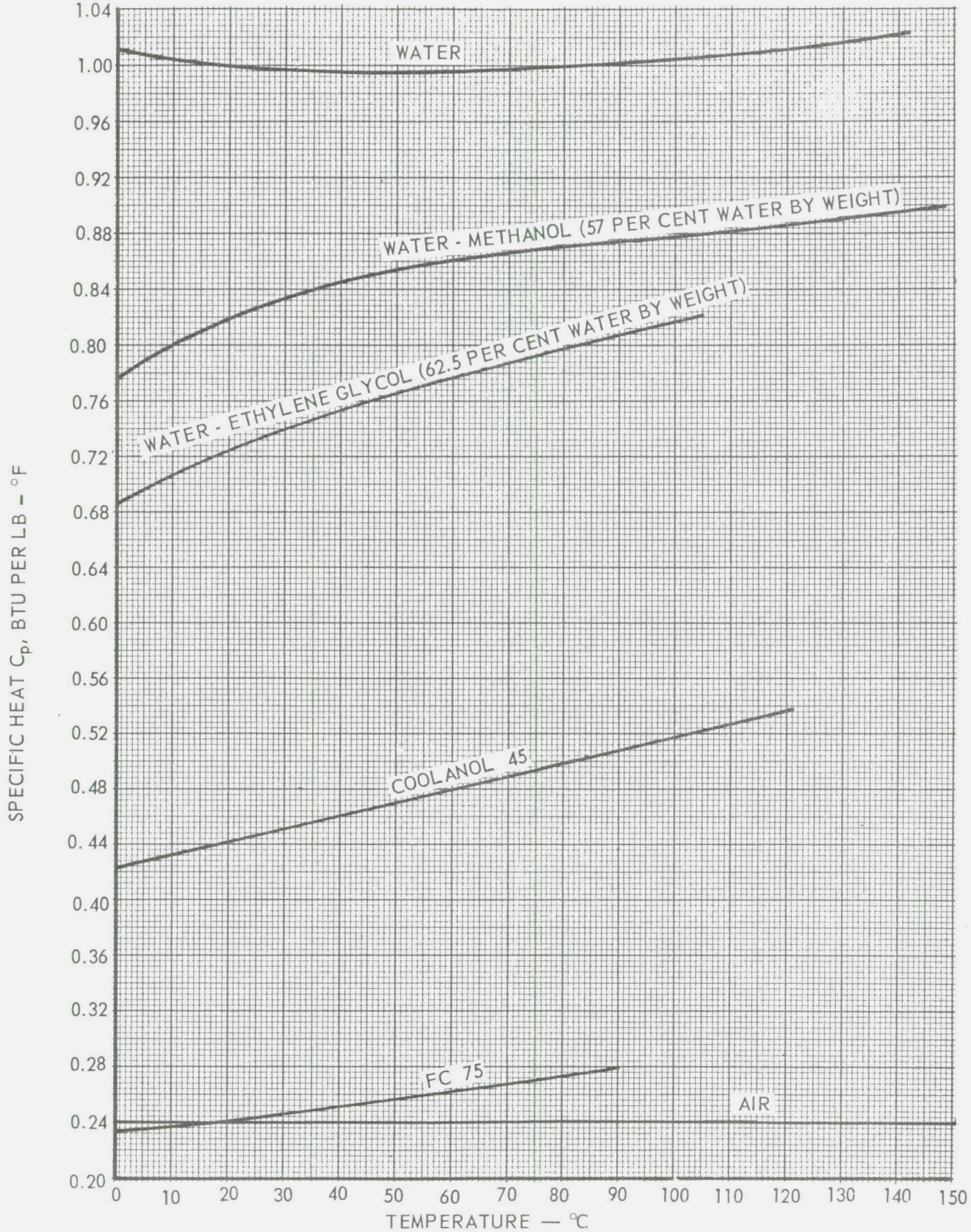


Figure 16–5 SPECIFIC HEAT OF HEAT TRANSPORT FLUIDS

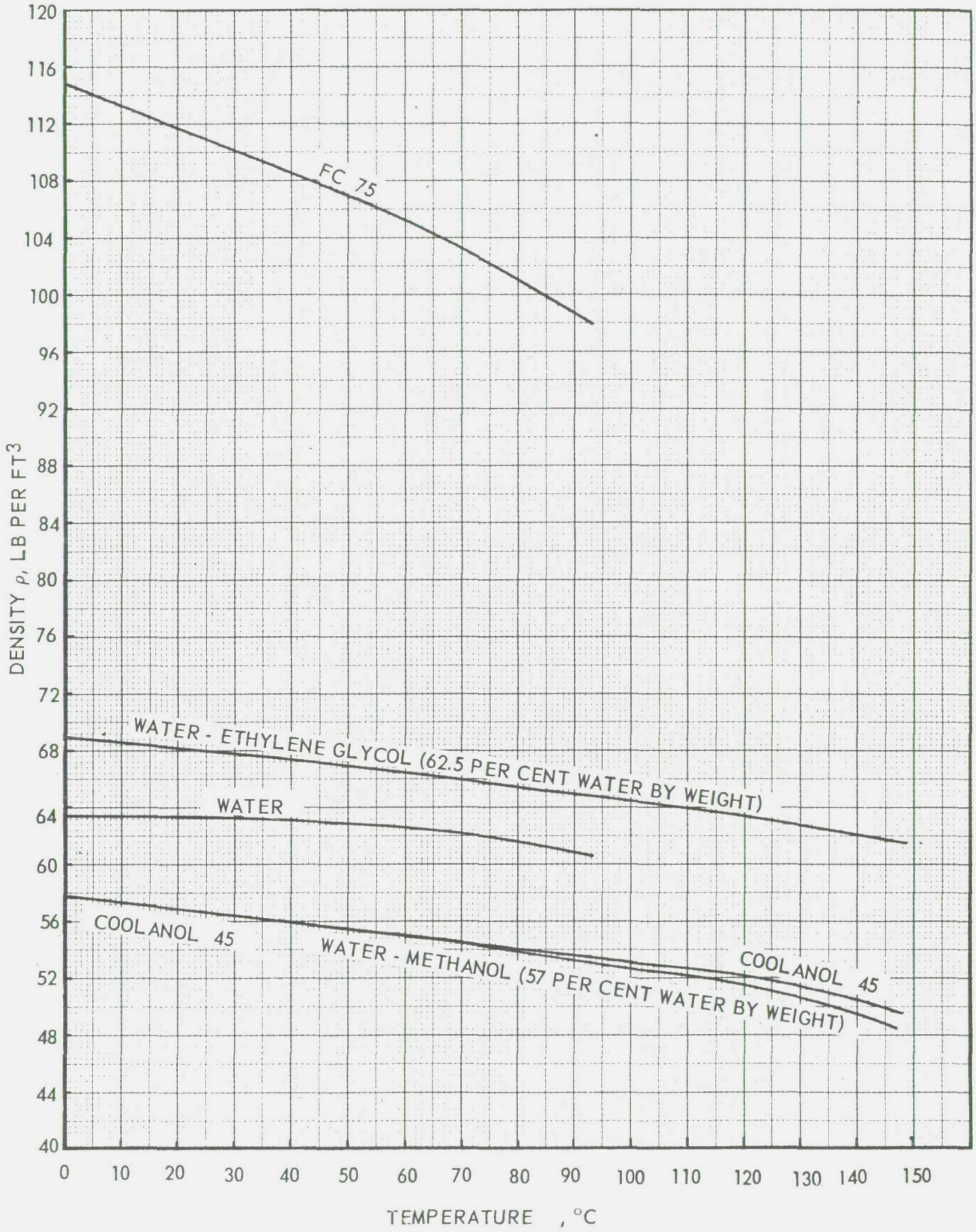


Figure 16–6 DENSITY OF HEAT TRANSPORT FLUIDS

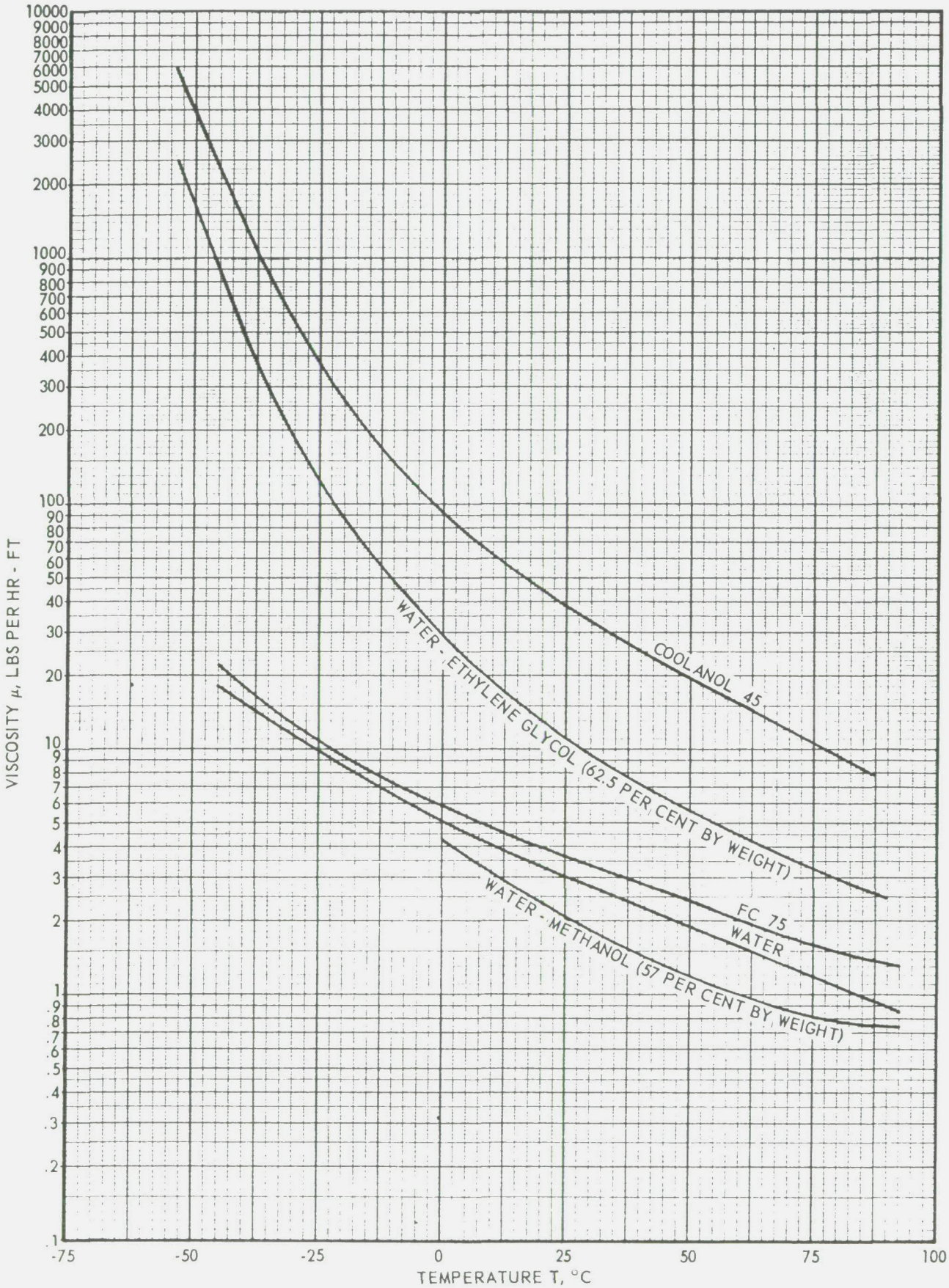


Figure 16–7 VISCOSITY OF HEAT TRANSPORT FLUIDS

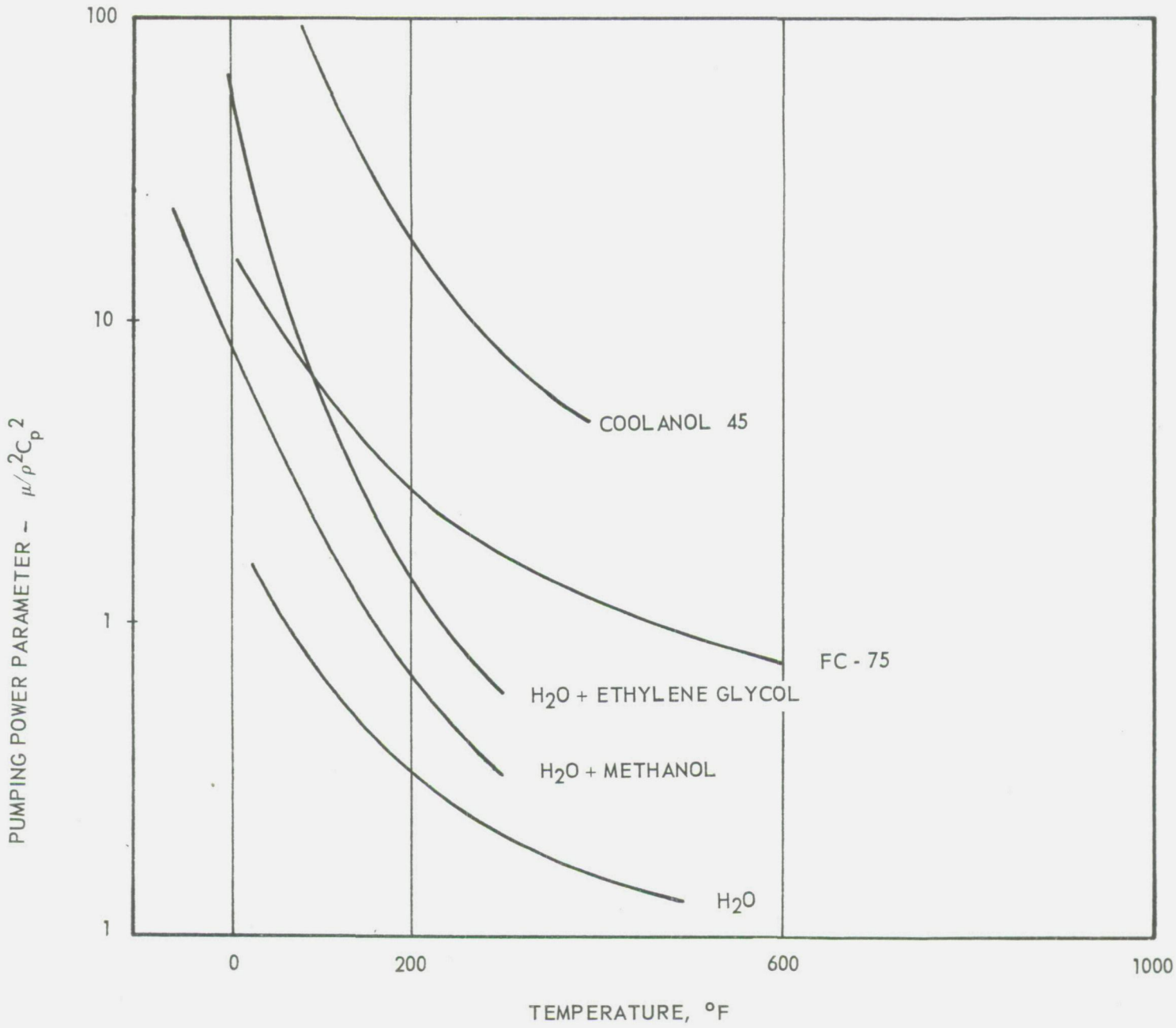


Figure 16–8 COMPARATIVE PUMPING POWER FOR HEAT TRANSPORT FLUIDS (LAMINAR FLOW)

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

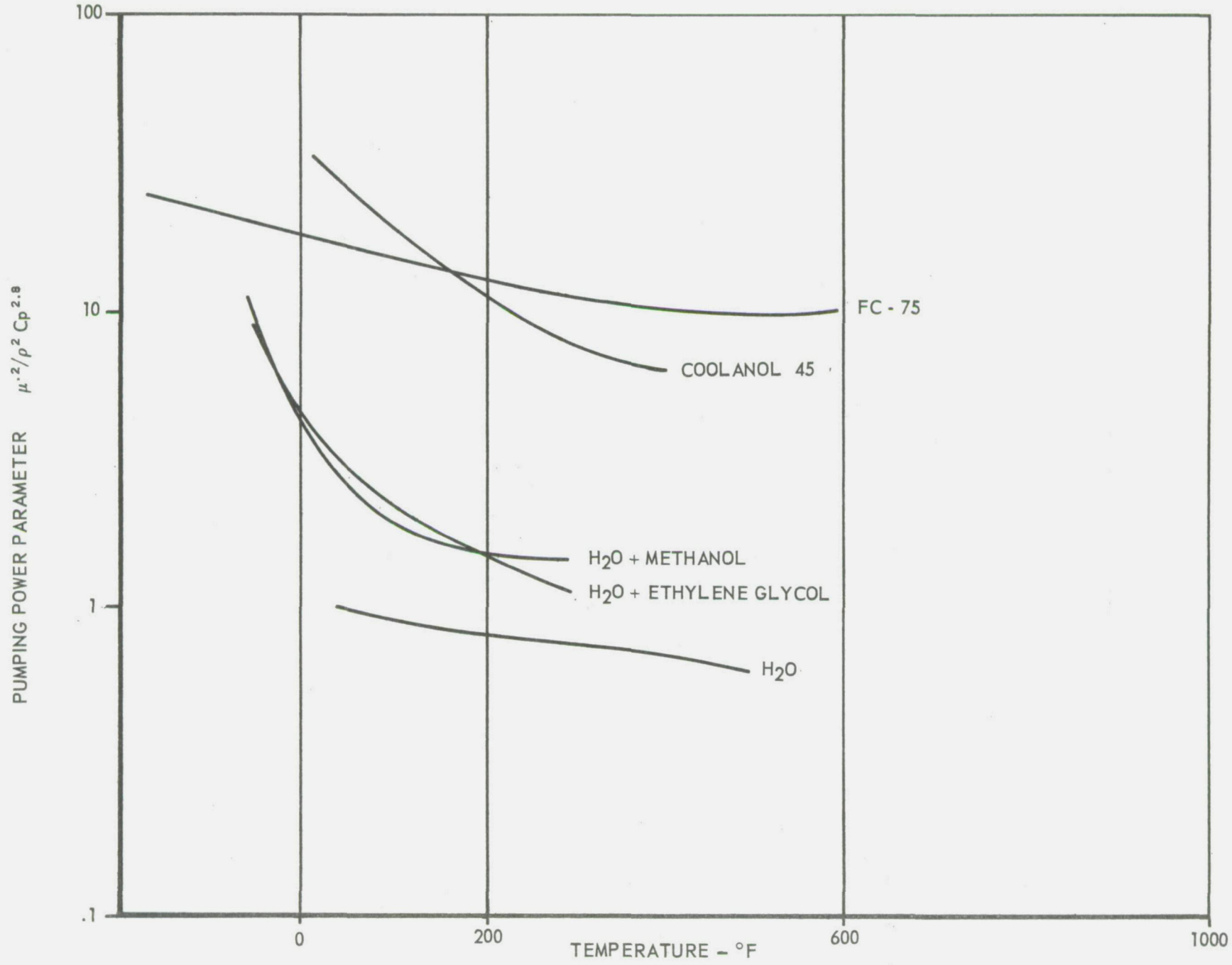


Figure 16–9 COMPARATIVE PUMPING POWER FOR HEAT TRANSPORT FLUIDS (TURBULENT FLOW)

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

is approximately 10% of that required for dielectric fluids (FC-75, coolanol 45, etc.).

Water solutions have the additional advantages of being relatively non-toxic and being suitable for use as an expendable heat sink. However, the freezing temperature of pure water requires inclusion of an additive for freezing point depression. Good results have been accomplished in this area with ethylene glycol as an additive. A 62.5% glycol 37.5% water solution begins to "slush" at about -50°F. This is well below the expected range of transport fluid temperatures and provides a safe margin against fluid freezing and pump failure. Therefore, a 62.5% ethylene glycol - 37.5% water solution is recommended as the heat transport fluid for the LEM.

16.3.2.5.3 Heat Dissipation

The practical methods available for dissipation of the vehicle heat load to space are boiling of an expendable coolant and space radiators. Final selection requires consideration of the duration of each mission phase and the various penalties associated with each method. Space radiators have high fixed weight penalties per unit of heat dissipated with little or no additional weight penalty for increased time. Conversely, expendable evaporants have low fixed weights and high time-weight penalties. The LEM maximum heat loads occur during the short duration phases of lunar orbit, lunar landing and launch. The major portion of the LEM mission is spent at rest on the lunar surface under relatively low heat load conditions. A space radiator capable of dissipating the maximum load is prohibitive in weight and volume, and the quantity of expendable evaporants is prohibitive for extended lunar rest times. A system incorporating both methods can be optimized to obtain a lighter total weight than is possible using either system alone. Therefore, the recommended LEM heat dissipation method is a combination consisting of an expendable evaporant system and a space radiator.

16.3.2.6 Thermal Control Outside Pressure Shell

The equipment and structure outside the pressure shell will be subjected to extremes in thermal environment. Temperatures as low as -200°F may occur on thermally isolated components during lunar cold side landings. The temperatures during translunar flight, lunar orbit and lunar hot side rest depend on the degree of illumination by the sun, geometry, and surface radiative properties.

In order to maintain the equipment outside the pressure shell within acceptable temperature limits, special thermal control procedures will be required. The majority of the external items can be controlled by passive thermal control coatings and insulation. Thermal gradients can be greatly reduced by low emissivity coatings such as polished aluminum or gold. The low emissive coatings also reduce the heat loss by components in the cold environment. Such items as surfaces of cryogenic tanks should be kept as cold as possible. These can be painted with a low solar absorptivity-to-emissivity ratio white paint (e.g., titanium dioxide, TiO_2 base paint). Because of the high vacuum of outer space, "super-insulations" can be used to reduce the heat transferred to or from any of the components without paying a large weight penalty.

Electrical heaters can be employed where the equipment cannot be adequately controlled by passive means and only moderate heater power is required. The propellant lines to the reaction control thrust chambers are an example where this type of control is required. Any electrical equipment that dissipates significant heat and requires close thermal control can be mounted on the water-glycol cold-plates. The water-glycol from the thermal control loop can be passed through these cold-plates before entering the radiator.

The exact temperature limits of the structure and components will only be determined after a detailed design has been performed. The specific method of control will be defined by the temperature limitations and geometry. Further study is required to define specifically the external thermal control methods for all items.

16.3.2.7 Air Lock Trade Study

In order to provide the crew with the capability to explore the lunar surface, provisions for ingress and egress from the spacecraft must be provided. The following methods were examined to determine the best approach for the LEM.

(a) No air lock - cabin atmosphere dumped for ingress and egress.

(b) Air lock - air lock atmosphere dumped for ingress and egress.

(c) Vacuum system provided to pump the atmosphere into a storage receiver.

Figure 16-10 is a weight trade curve of methods (a) and (b) above. This figure is based on the use of a supercritical oxygen storage system, and shows that for a 400 ft.^{3} cabin an air lock is lightest if more than 4 cabin depressurizations are required. If a gaseous oxygen storage system is used, the storage weight penalty increases and an air lock becomes lighter at an even lower number of depressurizations.

A brief investigation of the possibility of using a vacuum pumping system to conserve the atmosphere was conducted. The disadvantages associated with this method are as follows:

(a) Contamination of the air with existing mechanical type vacuum pumps.

(b) High power requirements.

(c) Long pumpdown times.

(d) Unavailability of light weight reliable components.

In view of the above disadvantages the pumping technique appears to be unsuitable.

Based on the trade data of Figure 16-10, use of an air lock is the lightest method for ingress and egress from the LEM. Approximately 2 to

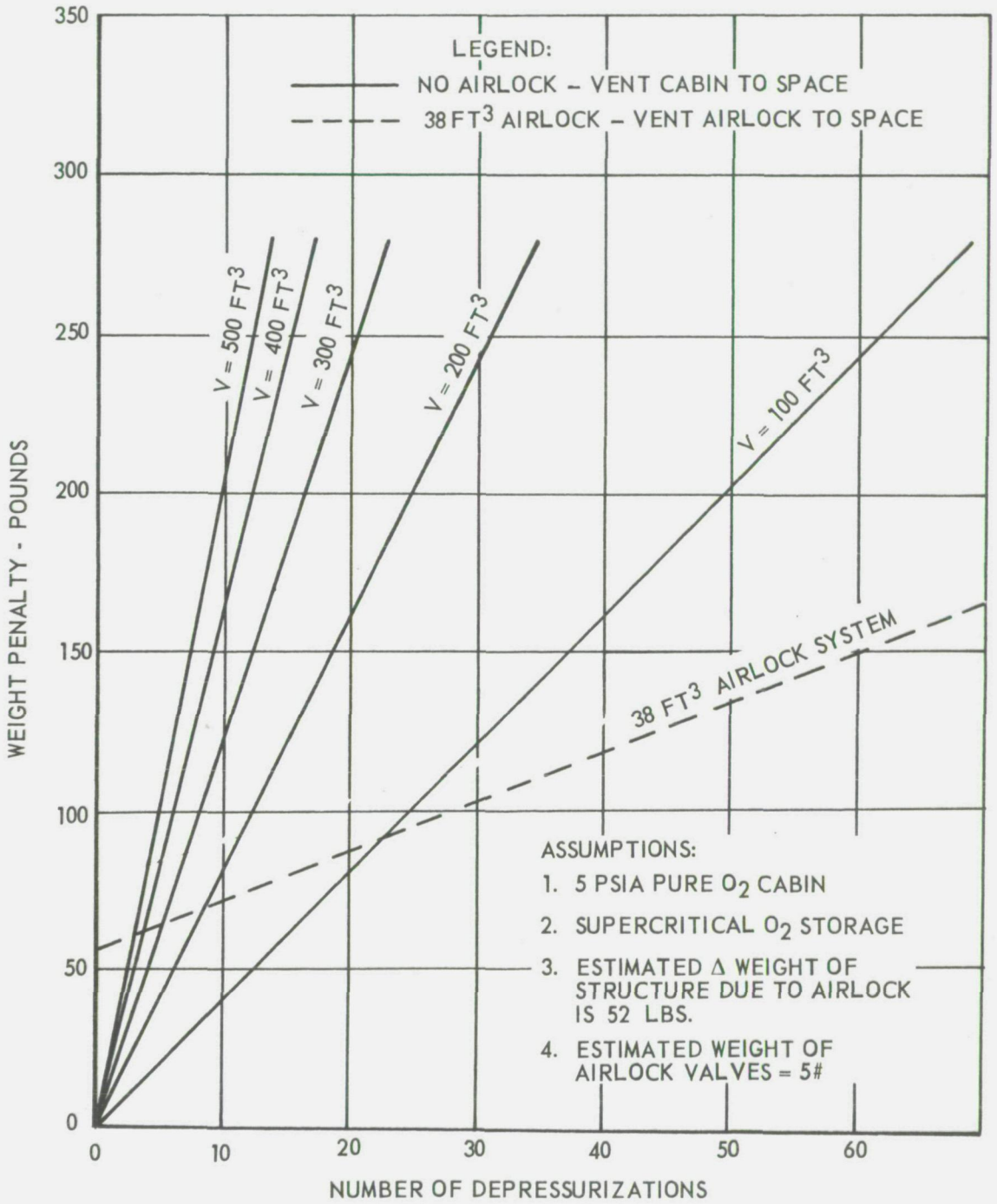


Figure 16–10 WEIGHT PENALTY FOR INGRESS & EGRESS FROM PRESSURIZED CABIN

4 trips outside the vehicle are planned for a 1-day lunar stay time. The capability for additional trips is required for longer lunar stay times and for ingress and egress from the LEM during the translunar phase of the mission. Therefore, an airlock has been incorporated in the recommended vehicle design.

16.3.2.8 Translunar Operation

The use of the Apollo service module propulsion system for mid-course velocity correction requires that the LEM be transferred to the front of the command module early in the flight. Since the LEM has a large free volume, it provides an excellent area for relaxation during the translunar portion of the mission. The large window area in the LEM also makes it desirable as a position of observation.

Provision of a shirtsleeve environment in the LEM during the translunar flight imposes operational and weight penalties on either the LEM or the Apollo environmental control systems. Conditioning of the atmosphere by the LEM environmental control system requires that the system be operated for a longer period of time. This additional time requires more expendables and increases the LEM storage weight penalties. This results in an increase in the LEM propulsion system requirements thereby causing a large increase in total vehicle weight. The additional expendables and storage provisions can be added to the Apollo command module with less penalty to the total vehicle. Therefore, the LEM environmental control system should be inactive and the entire vehicle should be conditioned by the command module ECS. This allows the crew to utilize the LEM space during the translunar phase at a minimum penalty to the overall mission.

In order to provide the above arrangement, an adapter-connector is required between the LEM and command module airlocks so that transfer of crew members can be made through a pressurized unit. The tunnel connections to the command module and the LEM present a serious leakage problem if the tunnel is pressurized throughout translunar flight. An alternative would be to pressurize the tunnel during crew transfer and allow the tunnel to evacuate after transfer. A comparison of the air lost by tunnel leakage is shown for both of these methods in Figure 16-11. The method selected depends on the tunnel leakage rate obtainable and the number of crew transfers between vehicles. If a low leakage seal can be developed, the best method is to maintain the tunnel pressurized at all times. This would permit unhampered crew transfer between the two vehicles. Figure 16-12 shows the weight penalty to the command module for translunar occupancy of the LEM. The weights given do not include the weight of the connector-adapter between the two vehicles. If a detailed system analysis indicates additional blowers are required, the LEM cabin fans can be operated. This presents no major problem since an electrical connection is already required for lighting in the LEM. The fan power requirements are not excessive.

16.3.2.9 Fuel Cell - Environmental Control System Integration

16.3.2.9.1 Integration of Cryogenic Stores

Since both the fuel cells and environmental control system use oxygen which is stored supercritically, integration of the storage tanks

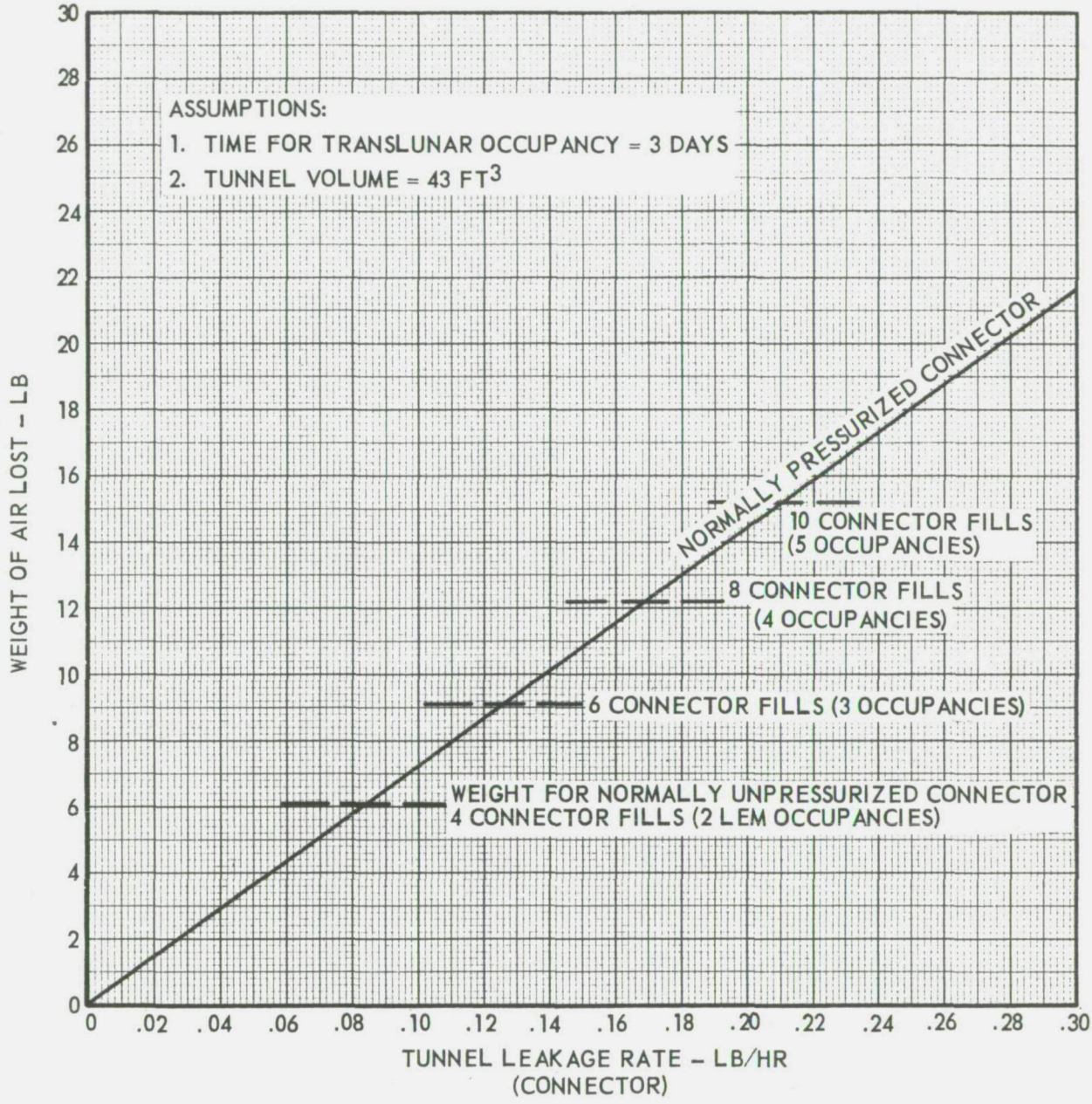


Figure 16–11 AIR LOSS FOR PRESSURIZED AND UNPRESSURIZED LEM – APOLLO CONNECTING TUNNEL

ASSUMPTIONS:

1. 2 ROUND TRIPS TO LEM PER DAY
2. LEM LEAK RATE = .1 LB/HR
3. UMBILICAL LEAK RATE = .05 LB/HR
4. CONNECTION PASSAGE VOLUME = 43 FT^3

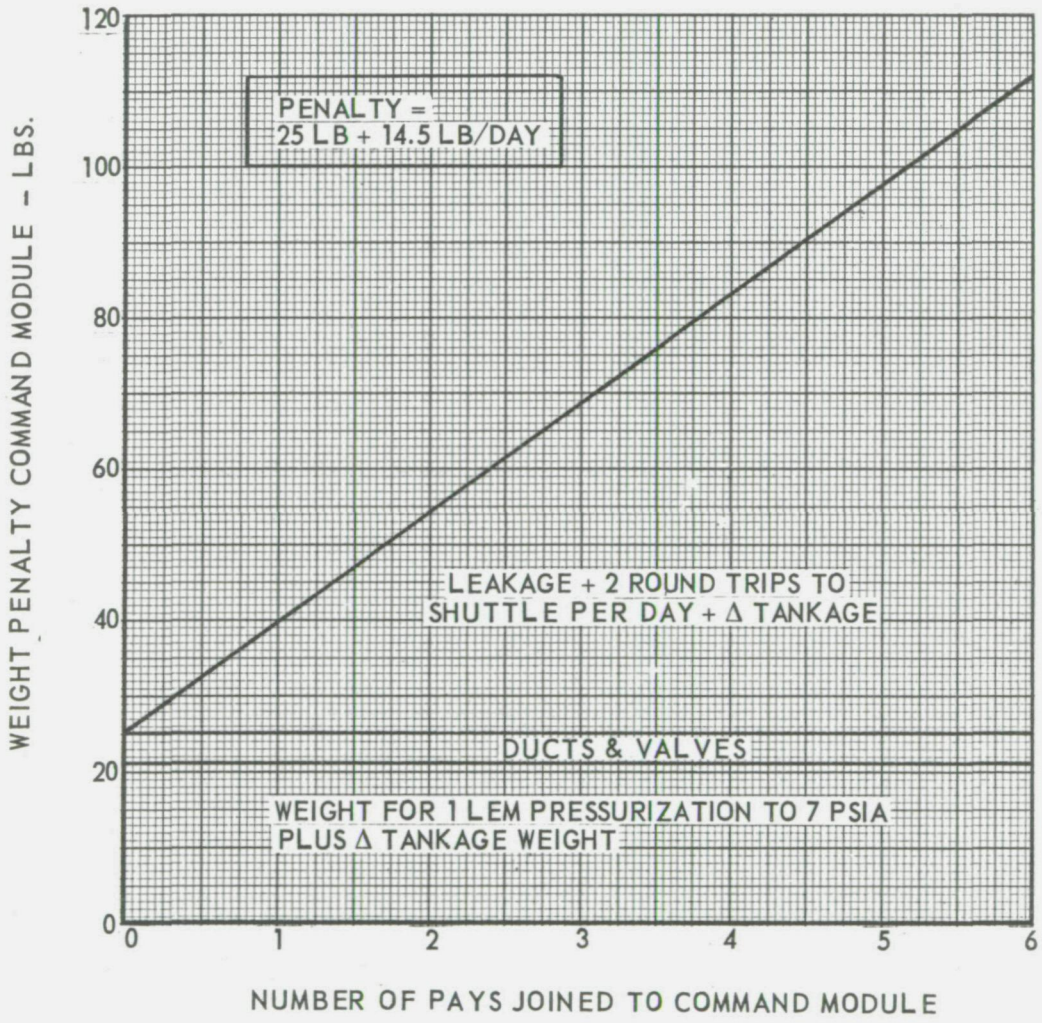


Figure 16–12 WEIGHT PENALTY TO COMMAND MODULE FOR TRANSLUNAR OCCUPANCY OF LEM

appeared feasible. A combination of the oxygen supplies was investigated to determine the advantages and disadvantages of integration. Figure 16-13 shows a schematic of the integrated and non-integrated oxygen storage systems which were compared. With the integrated concept, oxygen is supplied to the fuel cells and the ECS from either of the two oxygen tanks. The non-integrated system utilizes two tanks for the fuel cells and a single separate tank for the ECS.

A comparison of these two systems showed a negligible weight difference (approximately 2 lbs.). In addition, the reliability of the non-integrated concept is slightly better. Separate oxygen supplies eliminate the interdependence of the fuel cell and ECS allowing greater flexibility. Therefore, the non-integrated concept was selected.

16.3.2.9.2 Thermal Control Integration

The excess heat and water vapor generated in the fuel cell are removed by re-circulation of hydrogen. (See Volume IV, Section 9.0 for a description of the fuel cell.) This hydrogen must be cooled externally. The two most feasible ways of cooling the hydrogen and condensing excess water are (1) passage through a separate space radiator and (2) passage through a heat exchanger in the ECS water-ethylene glycol thermal control loop (glycol loop). The former method is a non-integrated concept and the latter involves integration of the fuel cell and the ECS. Figure 16-14 presents schematics of these two methods.

The approximate limiting fuel cell heat loads and temperature ranges are as follows:

	Minimum Power (Lunar Rest)	Maximum Power (Lunar Orbit, Landing, Launch)
Heat Load	2300 BTU/hr	7100 BTU/hr
Water vapor exit temperature from fuel cell	180°F	485°F
Hydrogen temperature at cooling system exit - fuel cell interface	35°F min.	141°F max.

A separate fuel cell radiator designed to operate on the cold side of the lunar surface without freezing condensed water is not capable of rejecting the required heat load in lunar orbit (i.e., 7100 BTU/hr as shown in the above summary). In lunar orbit, the radiator is heated by solar radiation, radiation from the lunar surface, and the heat load from the fuel cell. Therefore, for a separate fuel cell radiator system, a water boiler must be included to augment the radiator cooling capacity under the maximum conditions of lunar orbit and lunar hot side operations.

Integration of the fuel cell cooling with the ECS thermal control loop involves the addition of a hydrogen-to-glycol heat exchanger and increasing the cooling capacity of the glycol loop. With this method, the minimum

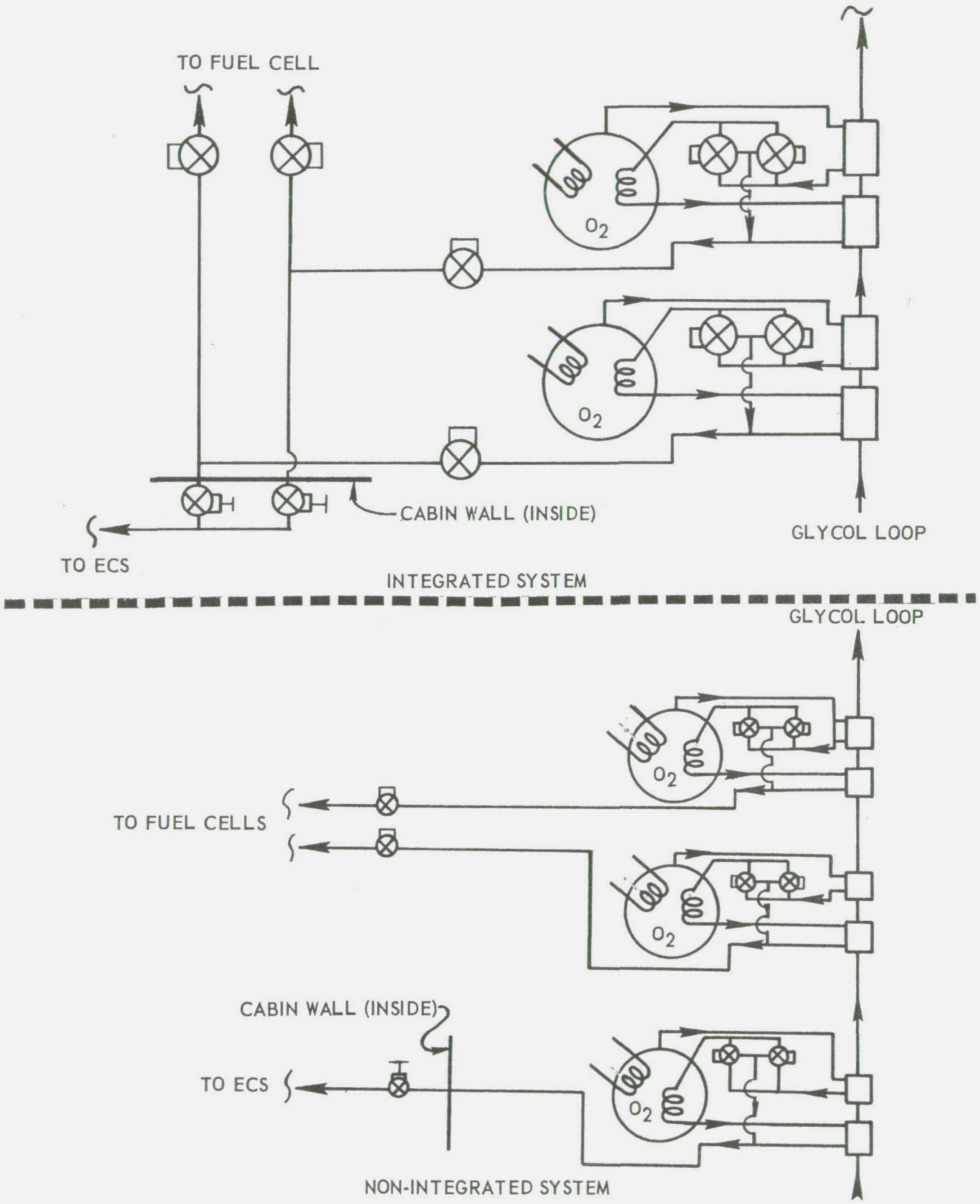


Figure 16–13 SCHEMATICS FOR INTEGRATED AND NON-INTEGRATED FUEL CELL AND ECS CRYOGENIC STORAGE

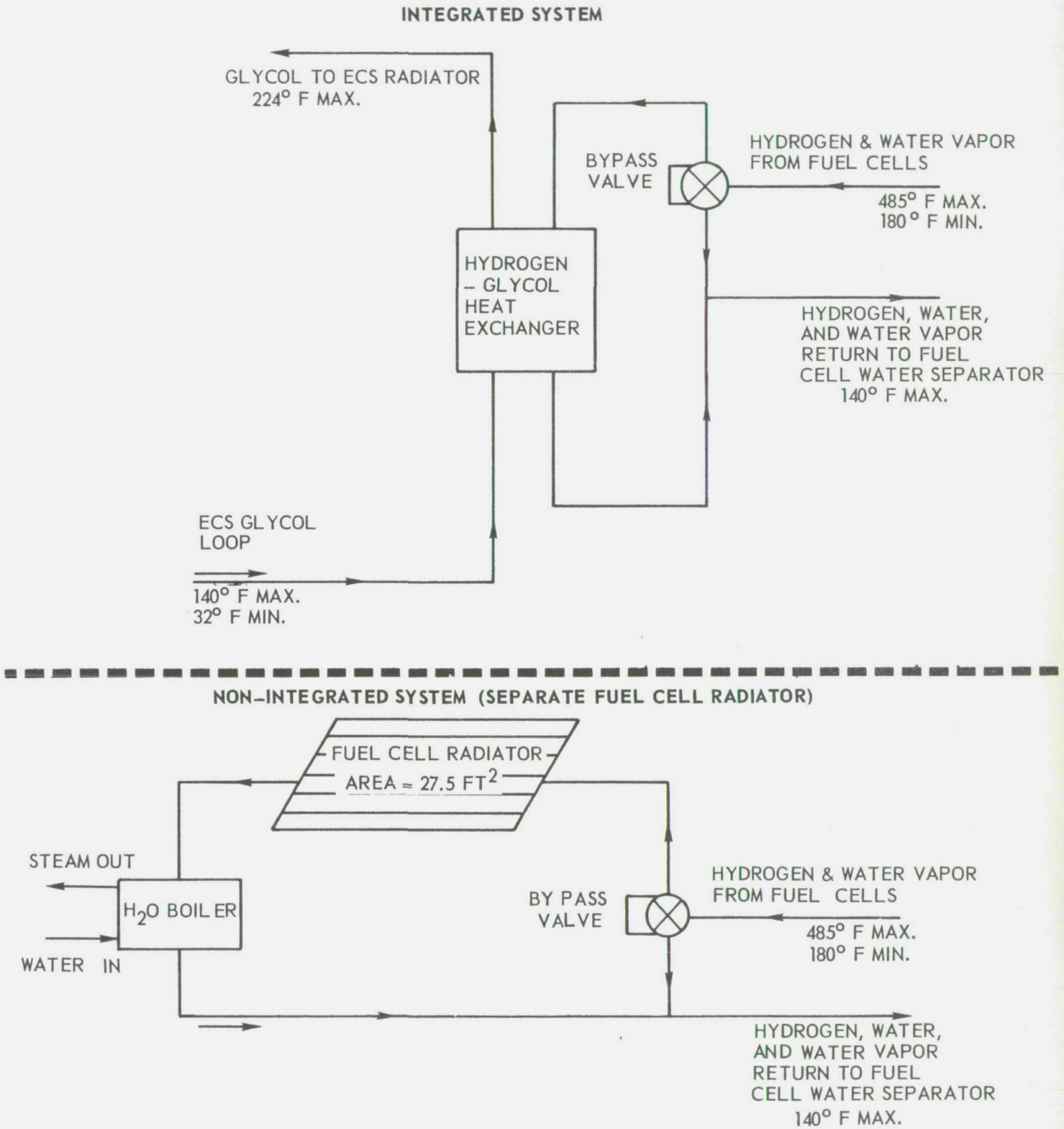


Figure 16–14 SCHEMATICS OF INTEGRATED AND NON–INTEGRATED FUEL CELL AND ECS THERMAL CONTROL LOOPS

glycol inlet temperature to the fuel cell heat exchanger would be approximately 125° F. (counter flow heat exchanger), therefore, no freezing problem would be encountered. The hydrogen from the fuel cells can be cooled adequately by adding the heat exchanger in the glycol loop between the cryogenic expulsion heat exchangers and the radiator inlet (Figure 16-14). With this system the maximum glycol outlet temperature is 207°F. No additional glycol loop controls are required for the hydrogen cooling. The glycol radiator size and the coolant water quantity must be increased for this integrated concept. For lunar cold side operation, the increase in radiator size is approximately 17 ft^2.

A comparison of the two concepts indicates that the weight of providing a separate radiator, water boiler and controls is approximately the same as the weight obtained by integrating the fuel cell cooling function with the ECS. Since a separate fuel cell radiator system would require an additional water boiler and its associated controls, thereby reducing reliability, the concept of integrating the fuel cell cooling into the thermal control loop was selected.

16.3.2.9.3 Water Management

Since the fuel cell produces pure liquid water as a by-product, this water can be used as drinking water or as an expendable coolant. The fuel cells produce approximately 0.9 lbs. of water per kilowatt-hour of electricity generated. An accumulator in the fuel cell system collects the condensed water and separates this water from the fuel cell hydrogen coolant flow. This accumulator has a storage capacity for 24 hours of fuel cell operation. During lunar rest, which accounts for the major part of the LEM operational mission, approximately 1.2 lb/hr of water is produced. This is more than adequate to supply drinking and wash water for a 2-man crew. The excess water produced is used as cooling water during the rendezvous phase of the flight and/or augmentation of the ECS radiator during lunar rest on the hot side. A schematic of the water management loop of the ECS is included in Figure 16-40.

During extended stays on the lunar cold side, more water is produced by the fuel cells than is required for crew consumption and cooling. This excess water is stored in mylar bags, sealed, and left on the lunar surface prior to launch. Storage of the water in the LEM would impose an unnecessary penalty on the lunar launch propulsion system.

16.3.2.10 Waste Handling

Off loading urine, wash water and other waste products on the lunar surface or in lunar orbit present the possibility of contamination of the lunar environment. During the Apollo spacecraft studies, the decision was made to design the command module with the capability to return these waste materials to earth. Further study was recommended to determine the extent of the contamination problem and the feasibility of leaving the wastes on the lunar surface or in space.

In keeping with the Apollo decision, the urine and wash water from the LEM can be stored throughout the mission and transferred to the command module. The waste may then be returned to earth. A significant

weight reduction in the LEM can be obtained by leaving these wastes on the lunar surface prior to lunar launch. Table 16- X shows the weight penalties to the command module and the LEM due to return of the waste. Additional studies are needed to determine methods of decontamination which would permit waste and wash water to be left on the lunar surface.

16. 3. 2. 11 Reliability Philosophy

In general, the inherent reliability of a system is established by the basic design. A complicated system which has many operational parts, such as the environmental control system, may have a low overall reliability even if each part has an extremely high reliability. For a combination of functions, parts, etc., assuming each has equal importance, the following equations can be used to determine overall system reliability.

Series Combination

$$R = e^{-t\ (\lambda 1\ +\lambda 2\ +\lambda 3\ ----)}$$

Parallel (Redundant) Combination

$$R = 1 - (1-R)^2$$

Where

R = Reliability of a combination of components

t = Operating time per flight (hrs.)

λ = Failure rate = $\frac{1}{\text{MTBF}}$

MTBF = Mean time between failure = $\frac{1}{\lambda}$

For example, consider an atmosphere control system consisting of five components, arranged in series, with each component having equal importance and a mean time between failure of 10, 000 hours. The reliability for a 7-day mission is calculated below:

$$R = (17 \times 24)\left(\frac{5}{10,000}\right) = 0.92$$

The reliability for the same components arranged in two completely redundant systems is:

$$R = 1 - (1 - .92)^2 = 0.994$$

TABLE 16 – X WEIGHT PENALTY FOR RETURNING URINE AND WASH WATER TO EARTH

NO.	ITEM	WEIGHT PENALTY –LBS		
		24 HOUR LUNAR STAY	80 HOUR LUNAR STAY	160 HOUR LUNAR STAY
1	Δ WEIGHT OF APOLLO SPACECRAFT (WASTE WATER FROM THE LEM WHICH IS ADDED TO APOLLO FOR THE RETURN FLIGHT TO EARTH)	28	62	111
2	Δ WEIGHT OF LEM LUNAR LAUNCH PROPULSION SYSTEM AND ASSOCIATED EQUIPMENT FOR LAUNCH OF WASTE WATER FROM THE LUNAR SURFACE	42	93	166
3	Δ WEIGHT OF LEM FOR PORTABLE WASTE WATER STORAGE FACILITIES AND ADDITIONAL DESCENT PROPULSION CAPABILITY	70	177	318
4	Δ TOTAL WEIGHT OF LEM AT SEPARATION FROM APOLLO FOR LUNAR LANDING (SUM OF ITEMS 2 AND 3)	112	270	484

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

From the above, it is apparent that true system redundancy increases reliability. The reliability of the system can also be improved by providing redundancy in critical components. Since the mean time between failure of all components is not the same, more redundancy is required in some components (in general, dynamic components) than in others (static components).

Preliminary reliability analyses indicates that the required ECS reliability can be achieved by redundancy in components and by redundancy in systems. For preliminary system design and for trade studies, the following reliability philosophy was adopted.

(a) The failure of a single dynamic component will not affect the operation of the ECS

(b) Two failures of a dynamic component will require abort of the mission. The system will still have the capability for successful abort and rendezvous of the LEM with the command module.

(c) Static components, such as heat exchangers and ducts will be designed to have the same structural integrity as the basic vehicle structure and will require no redundancy.

In order to provide the reliability required for crew safety, separate redundant systems are provided for emergency thermal and atmosphere control. Portable life support units provide thermal and atmosphere control for the crew in the event of a failure of the basic environmental control system. Emergency cooling is provided for all essential electrical and electronic equipment by water boilers built as integral parts of the equipment cold plates.

Prior to final system design, a detailed reliability analysis of the entire ECS must be made. The actual or projected mean time between failure of the individual components will suggest some changes in the detailed arrangement. These changes, however, are not expected to alter the feasibility nor affect, to a significant extent, the weight of the selected environmental control system.

SYSTEM PARAMETRIC STUDIES

16. 3. 3 System Parametric Studies

Parametric analyses of the effects of crew size, mission duration, reliability philosophy and landing site were made in order to provide the ECS inputs required for over-all vehicle synthesis. These analyses required that a wide range of systems be examined. In order to cover this wide range, two basic systems, one a minimum pressure suit system and the other a combined pressure suit and shirtsleeve cabin system, were analyzed. A schematic of the minimum pressure suit system is shown in Figure 16-15. The combined pressure suit and shirtsleeve cabin system which was studied is essentially the same as the recommended LEM system which is discussed in Paragraph 6. 3. 5 and shown schematically in Figure 16-40. Additional equipment and expendables were added and removed from these two basic systems to determine: (1) weight vs time, (2) effect of crew size, (3) hot side landing weight penalty, and (4) the penalty for a reliability philosophy of single and dual dynamic component failures prior to abort.

Analyses to determine the effect of crew size and mission duration were made by separating the ECS components into fixed weight and time and/or crew variant weight components. System weight is then a summation of the fixed and variable weights for any mission duration and crew size. For simplicity of presentation, it was assumed that any mission requires sufficient ECS expendables to complete lunar touchdown and orbital rendezvous. Therefore, system weight is that required for landing and rendezvous plus that required for varying periods of lunar stay time. This results in a functional weight relationship of the form

$$W_t = (\text{fixed weight}) + (\text{weight for landing and rendezvous}) + (\text{weight for lunar rest})$$

or

$$W_t = \left[C_1 + C_2 \quad N\right] + \left[C_3 \, (N) \, t_f\right] + \left[C_4 \, (N) \, t_r + C_5 \, t_r\right]$$

where:

W_t = Total ECS weight

C_1 = Fixed weight which is independent of crew size or mission duration (e. g., pressure regulators).

C_2 = Fixed weight which is dependent on crew size (e. g., suit diverter valves).

C_3 = Expendable weight dependent on crew size and flight time (e. g., environmental O_2).

C_4 = Expendable weight dependent on crew size and lunar rest time (e. g., environmental O_2).

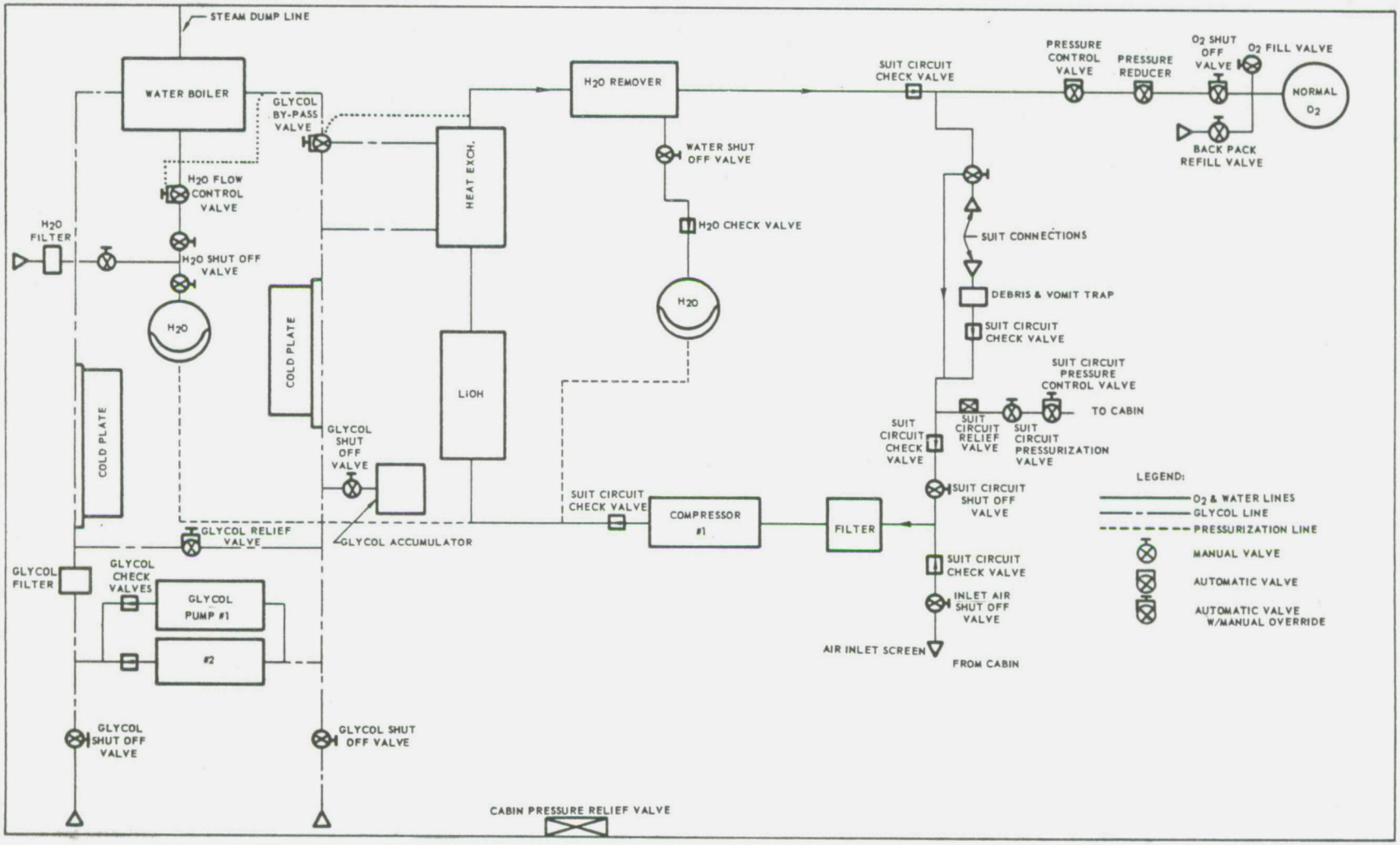


Figure 16–15 ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC (MINIMUM SPACE SUIT ONLY SYSTEM)

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

C_5 = Expendable weight which is essentially independent of of crew size but dependent on lunar rest time (e. g., expendable evaporants).

N = number of crew members

t_f = flight time

t_r = lunar rest time

The five constants (C's) in the above equation were determined from a detailed component breakdown for both the minimum pressure suit system and the combined pressure suit and shirt sleeve atmosphere system. The equation was then used to determine the weight variation in the two systems as a function of number of crew members and lunar stay time. Figure 16-16 presents a summary plot of the lunar cold side weights of each of these two systems.

The penalty for an unrestricted landing site (Δ weight for hot vs cold side) is due primarily to the additional vehicle heat dissipation system capability required. As shown in Figure 16-16 this weight penalty is approximately 125 lbs and 318 lbs for a 2 men - 24 hr and a 2 men - 160 hr system respectively. Figure 16-16 also shows the weight penalty for incorporating dual dynamic components in the minimum pressure suit system. This penalty is approximately 87 lbs.

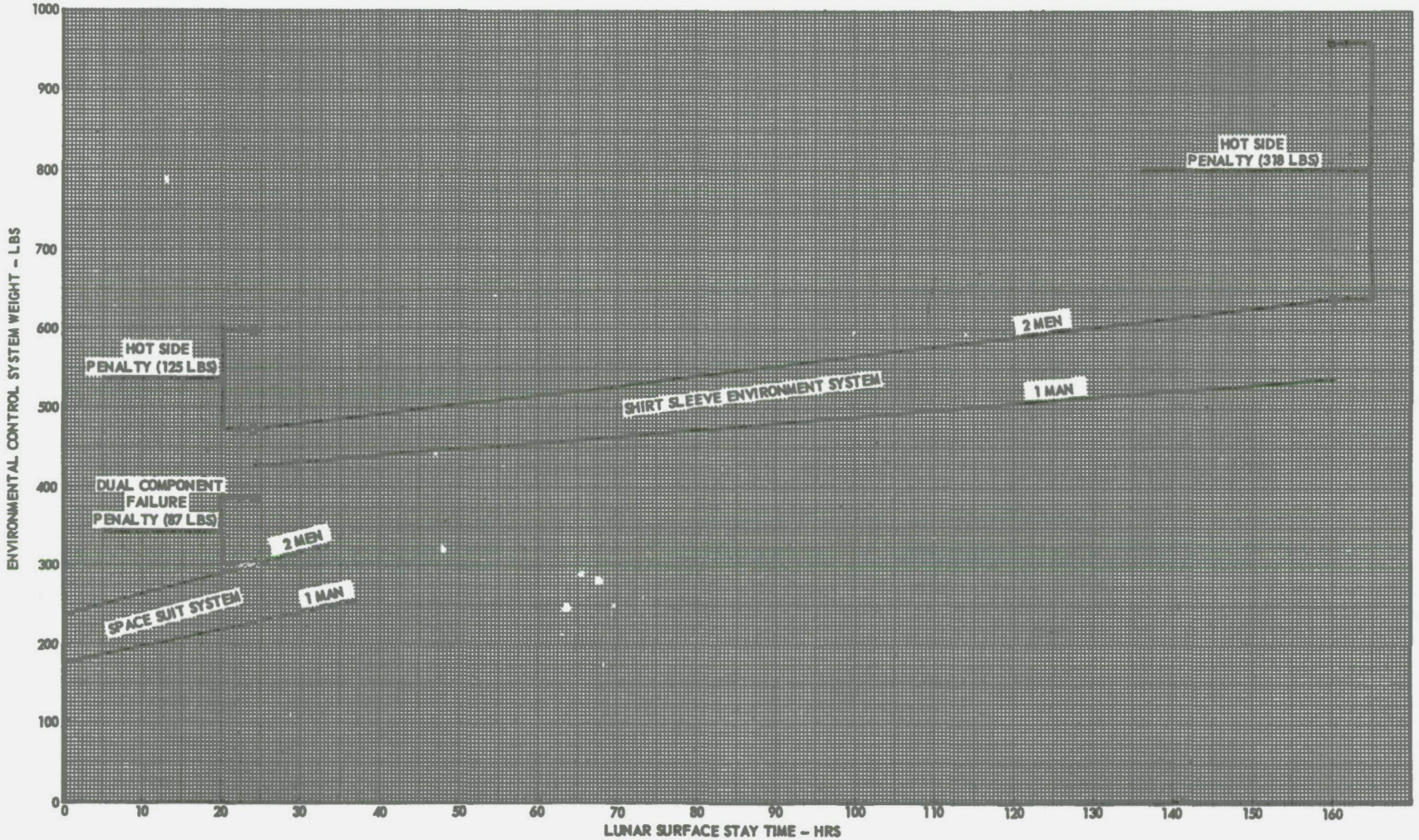


Figure 16–16 ENVIRONMENTAL CONTROL SYSTEM PARAMETRIC WEIGHT SUMMARY

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

ANALYSIS OF ENVIRONMENTAL CONTROL

16.3.4 Analysis of Environmental Control for the Selected Vehicle and Mission

16.3.4.1 Cabin Leakage

High cabin leakage rates impose a significant weight penalty on the vehicle. When the spacecraft cabin is pressurized, all hatches, connections, fittings, etc.,through the pressure shell present potential leakage problems. The air lock is of particular significance since airlock operation requires that doors be opened and resealed during flight. Leakage through emergency hatches, conduit penetrations, etc., can be minimized by effective sealing prior to flight.

Figure 16-17 shows the weight penalty imposed on a spacecraft due to cabin leakage. This figure accounts for both the oxygen gas lost and the additional tankage required to store the make-up gas. From Figure 16-17, it is apparent that high leakage rates cause significant weight penalties and that the LEM should be designed for a minimum practical leakage rate. Additional hours spent during manufacture to thoroughly check for leaks and to correct leakage areas will pay dividends in increased payload capability.

The study of Reference 9 indicates that design leakage rates of 0.15 lb/hr and 0.45 lbs/hr represent the practical minimum obtainable and maximum allowable respectively. Extrapolation of Mercury data indicates that 0.1 lb/hr is a reasonable design objective which can be attained with extra care in leak checking and sealing during manufacture. Therefore, a maximum leakage rate of 0.1 lb/hr has been selected as a design objective for the LEM. Because of the fact that leakage rates may increase due to vibration, a 50% excess of the normal oxygen requirements has been included in the vehicle stores. This safety factor should be adequate to replace unpredicted atmosphere losses which occur during flight.

Another consideration associated with leakage is the problem of a rapid cabin decompression. The ECS must be designed with the capability to supply sufficient make-up flow to maintain a cabin pressure of at least 3.5 psia for the length of time required for the crew to don pressure suits. Figure 16-18 presents the LEM cabin decompression times as a function of hole size for no inflow augmentation and for various oxygen make-up flow rates. With the recommended supercritical storage technique, the actual oxygen flow rate obtainable depends on the per cent of oxygen remaining in the tank. With the current LEM design, a flow rate of approximately 10 lbs/hr is available early in the mission when the oxygen tank is full and approximately 80 lbs/hr is available toward the end of the mission when the tank is almost empty. Additional inflow may be obtained if desired by opening the rendezvous O_2 tank shut off valve. (Figure 16-40).

16.3.4.2 Rocket Plume Expansion

The extremely low ambient pressures in space causes wide dispersion of the rocket exhaust plume. The expanded plume will affect the spacecraft by interaction with vehicle structural components such as the landing gear and by interaction with the lunar surface during landing. Test data presented in Reference 10 indicate that the plume boundary can be closely

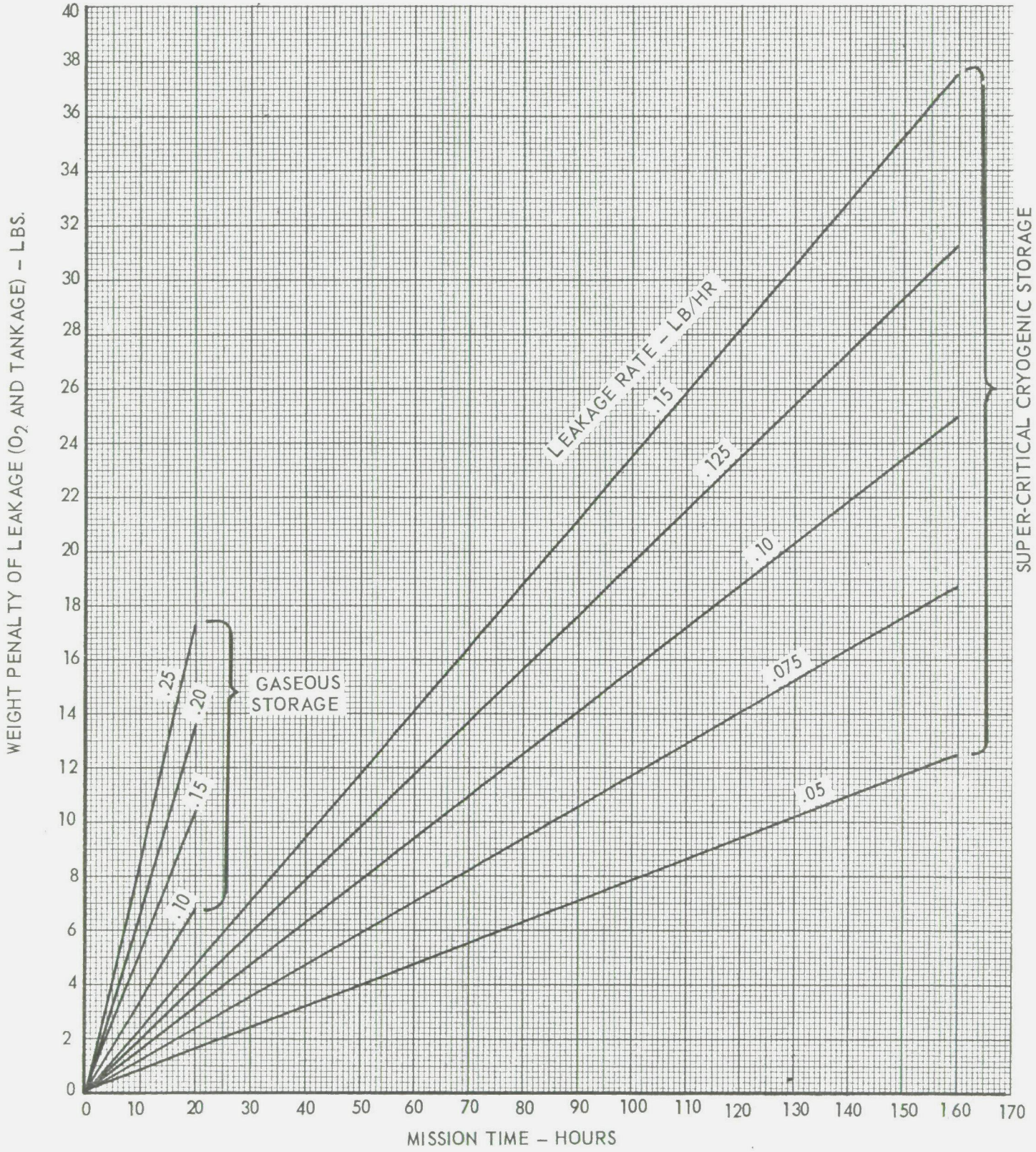


Figure 16–17 WEIGHT PENALTY FOR CABIN ATMOSPHERE LEAKAGE

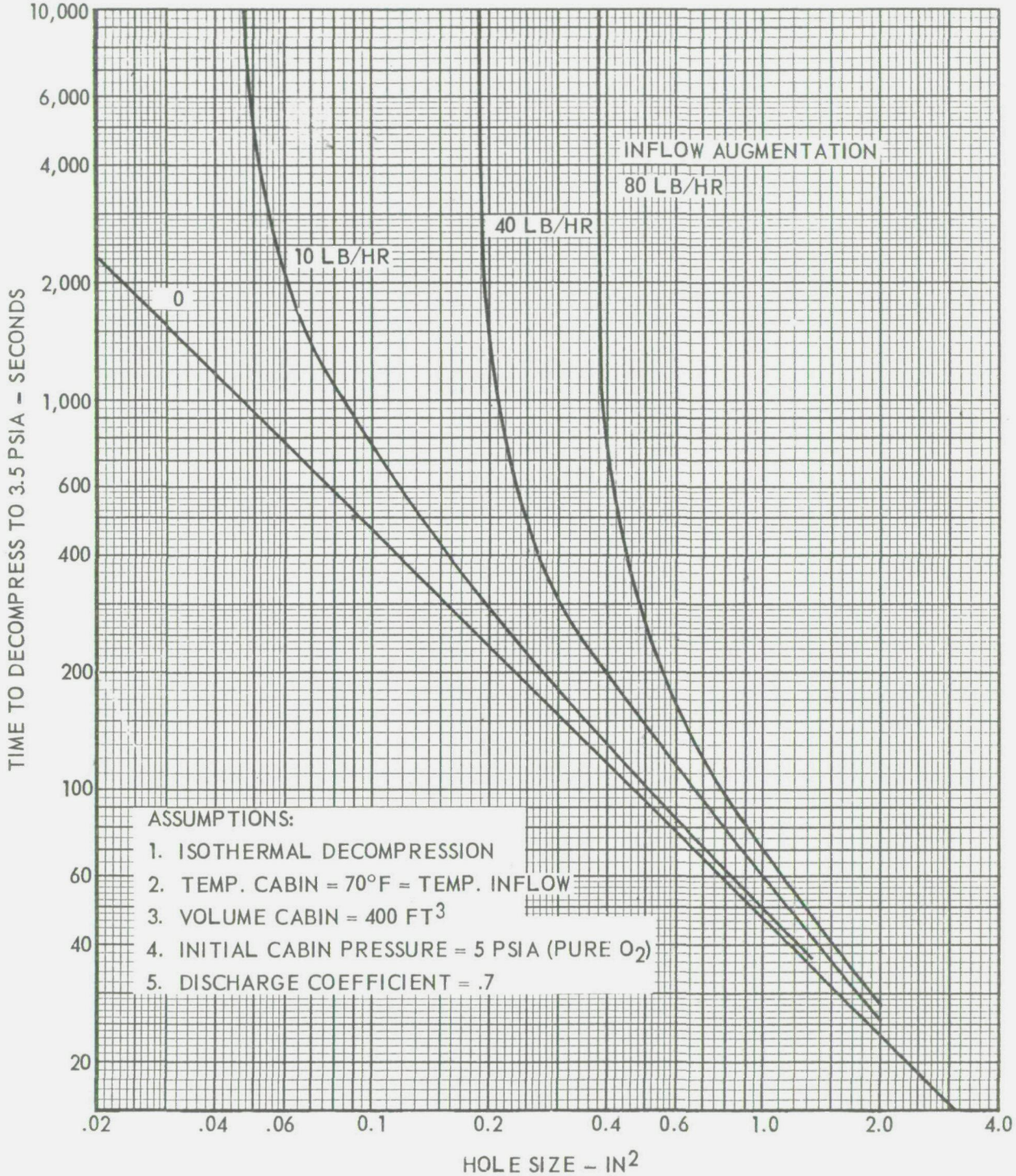


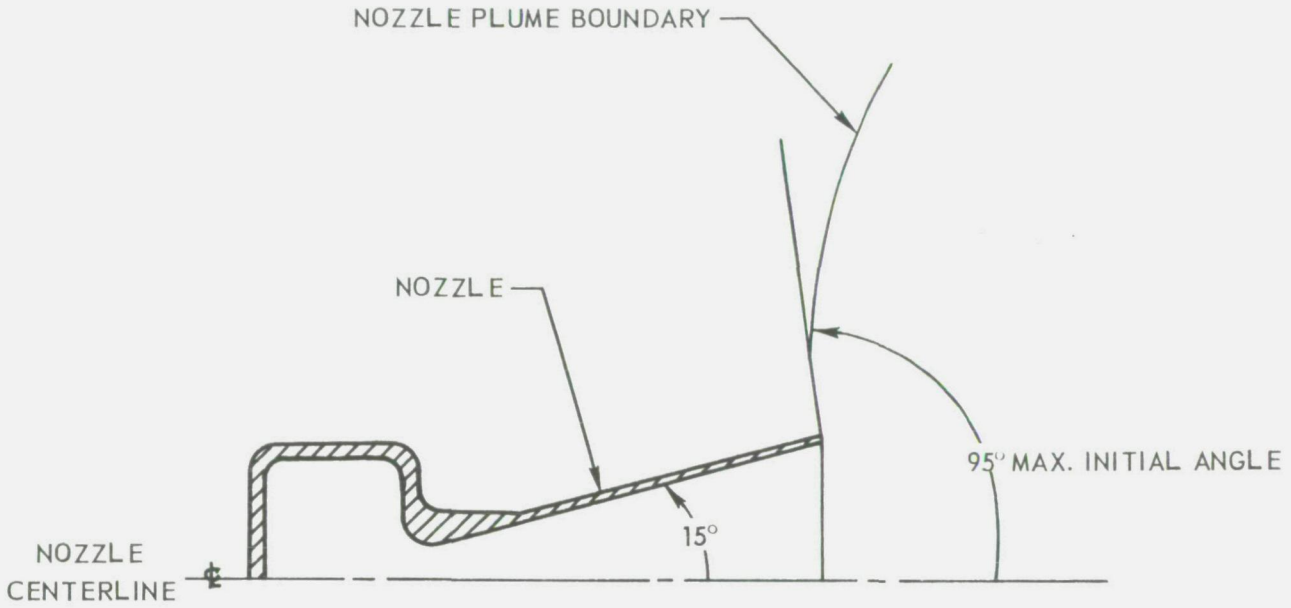
Figure 16–18 LEM CABIN DECOMPRESSION TIME

approximated by a circular arc. With the circular arc method, the initial angle of inclination of the flow is a function of the nozzle exit Mach number, nozzle exit half angle, ambient to stagnation pressure ratio, and the ratio of specific heats, γ . For a N_2O_4/Aerozine-50 rocket (γ =1.24) having an exit Mach number of 5.5 and a nozzle half angle of 15^o, the method of Reference 10 yields a maximum flow angle of approximately 95 degrees. This is illustrated in Figure 16-19. At extremely low pressures, gas dissociation and the change from continuum flow to free molecular flow may cause the plume boundary to deviate from the circular arc approximation. Estimated nozzle plume characteristics for a rocket motor burning N_2O_4/Aerozine-50 propellant at an ambient pressure altitude of 400 nautical miles are described in Reference 11. This pressure altitude corresponds roughly to the pressure of the lunar atmosphere. Isotherms, plume boundaries and the boundaries between continuum and slip flow are included in Reference 11.

Using the data of References 10 and 11, the effect of interaction between the nozzle plume and the LEM landing gear was estimated. The plume will not significantly affect other parts of the spacecraft. Figure 16-20 shows the maximum impingement of the exhaust plume on the landing gear. The plume provides heating in two ways; by radiation and by convection. Radiation does not impose a severe problem because of the relatively low exit static temperature and low emissivity of the plume. In addition, the radiant heating can be kept to a minimum by a gold coating on the landing gear structure.

Convective heating of the landing gear is much more difficult to analyze. Even though the static temperature of the gas may be relatively low as it expands downstream of the nozzle, much of the total temperature may be recovered when the gas impinges on the gear surface. Recovery, however, depends on the viscous properties of the gas (continuum flow). In free molecular flow, isentropic recovery does not exist and there will not be significant heating. Because the landing gear appears to extend only into the free molecular and slip-flow regions, no substantial recovery of temperature and pressure is expected.

As the nozzle plume impinges on the lunar surface during lunar landing, dust and loose layers of the surface will be eroded. Reference 12 presents the effects of a highly underexpanded jet impinging on a simulated lunar surface of white sand. These data indicate that lunar surface erosion will be a problem. Impingement of lunar surface dust on the vehicle may damage coatings and light structural materials. Reference 13 shows the flight paths of dust particles during a simulated lunar landing. These data indicate that a small amount of dust will impinge on the sides of the vehicle. During the firing of clustered nozzles, heavy impingement of dust may occur on the base area between the nozzles. This area will also be heated by backflow from the rocket nozzles and special considerations must be given to the design. Additional data on dust and particle impingement will be required before the lunar erosion problem can be assessed accurately.



NOTES:

1) PROPELLENT N_2O_4/AEROZINE 50 (γ = 1.24)
2) EXIT MACH NUMBER 5.5
3) AMBIENT PRESSURE ZERO (MAXIMUM EXPANSION)

Figure 16–19 **MAXIMUM NOZZLE PLUME EXPANSION (CIRCULAR ARC METHOD – REFERENCE 10)**

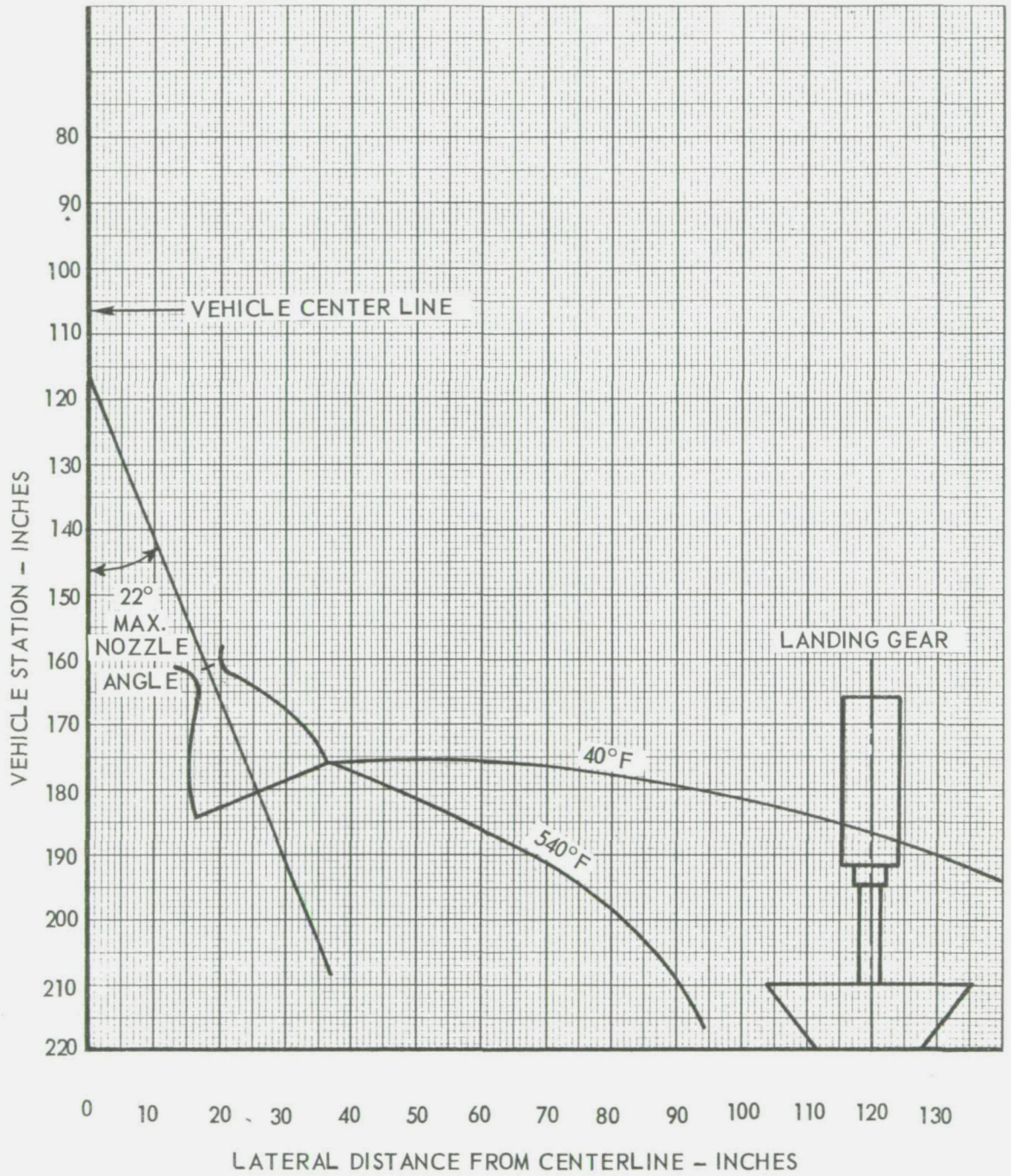


Figure 16–20 ESTIMATED NOZZLE PLUME TEMPERATURES IN VICINITY OF LANDING GEAR

16.3.4.3 Window Thermal Control

The complicated maneuvers of lunar landing and docking impose the requirement for good visibility. In order to achieve the required visibility, a large window area (approximately 46 ft.^{2}) is provided in the LEM. This window area imposes two general problems on the environmental control system. These are (1) condensation of water vapor on the glass and (2) an increase in heat leakage. Since the cabin atmosphere comes in contact with the glass (when the cabin is pressurized) the interior surface of the window must be kept above the dew point temperature in order to prevent condensation. On the hot side of the moon and in the sun a significant heat load will be added to the cabin through the glass. In a cold environment, heat will be lost by radiation. These problems are discussed in detail below.

The two extreme cold environments that will be experienced by the window are (1) cold side lunar landing and (2) orientation of the window to deep space. The lunar surface temperature may be as low as -250°F. At this lunar surface temperature, the vehicle receives essentially no radiant heating. The glass temperature depends on the heat transfer coefficient between the glass and the cabin atmosphere and the emissivity of the outer surface of the glass. Figure 16-21 shows the temperature of a window consisting of a single pane of glass for various outer surface emissivities and heat transfer coefficients.

Because of the low pressure in the cabin and zero or near-zero gravity fields, the heat transfer coefficient will be low. With moderate atmosphere circulation over the inside surface of the window, heat transfer coefficients of 1.0 to 1.5 BTU/hr.-ft.^{2}-°F. are probable. With these transfer coefficients, Figure 16-21 indicates that a surface emissivity of 0.15 to 0.20 is required to prevent condensation at a cabin relative humidity of 50%. Higher emissivities could be used, but excessive oxygen circulation rates over the inside surface of the window would be required to obtain heat transfer coefficients sufficiently high to prevent condensation. The total heat loss from the window would also be increased.

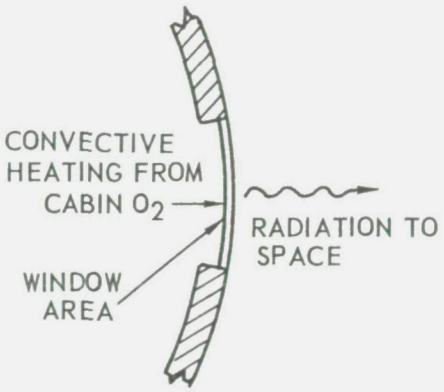
The heat transferred from the window can be decreased (and therefore higher window temperatures can be obtained) by using dual panes of glass separated by a vacuum gap. The glass temperatures and heat loss for this type of window are shown in Figure 16-22. Comparing Figures 16-21 and 16-22, it can be seen that the dual pane concept increases the window temperature significantly when the outer surface has a high emissivity. When the outer surface emissivity is low, however, the second pane has little effect.

Another method of keeping the window warm is to install electric heaters. With this method sufficient heat is added by the heaters to maintain the glass above the dew point of the atmosphere. The electrical power required for the heaters is shown in Figure 16-23 as a function of the inside surface heat transfer coefficient and the outer surface emissivity. As the heat transfer coefficient increases, less electrical power is required. A decrease in emissivity also decreases the electrical power requirements. The primary disadvantage of this method is the weight penalty imposed on the electrical power supply system.

CHANCE VOUGHT CORP

NOTES:

1) CABIN O_2 TEMPERATURE = 75°F
2) GLASS IS THERMALLY THIN
3) SPACE ENVIRONMENT TEMPERATURE = 0°R
4) ϕ = RELATIVE HUMIDITY



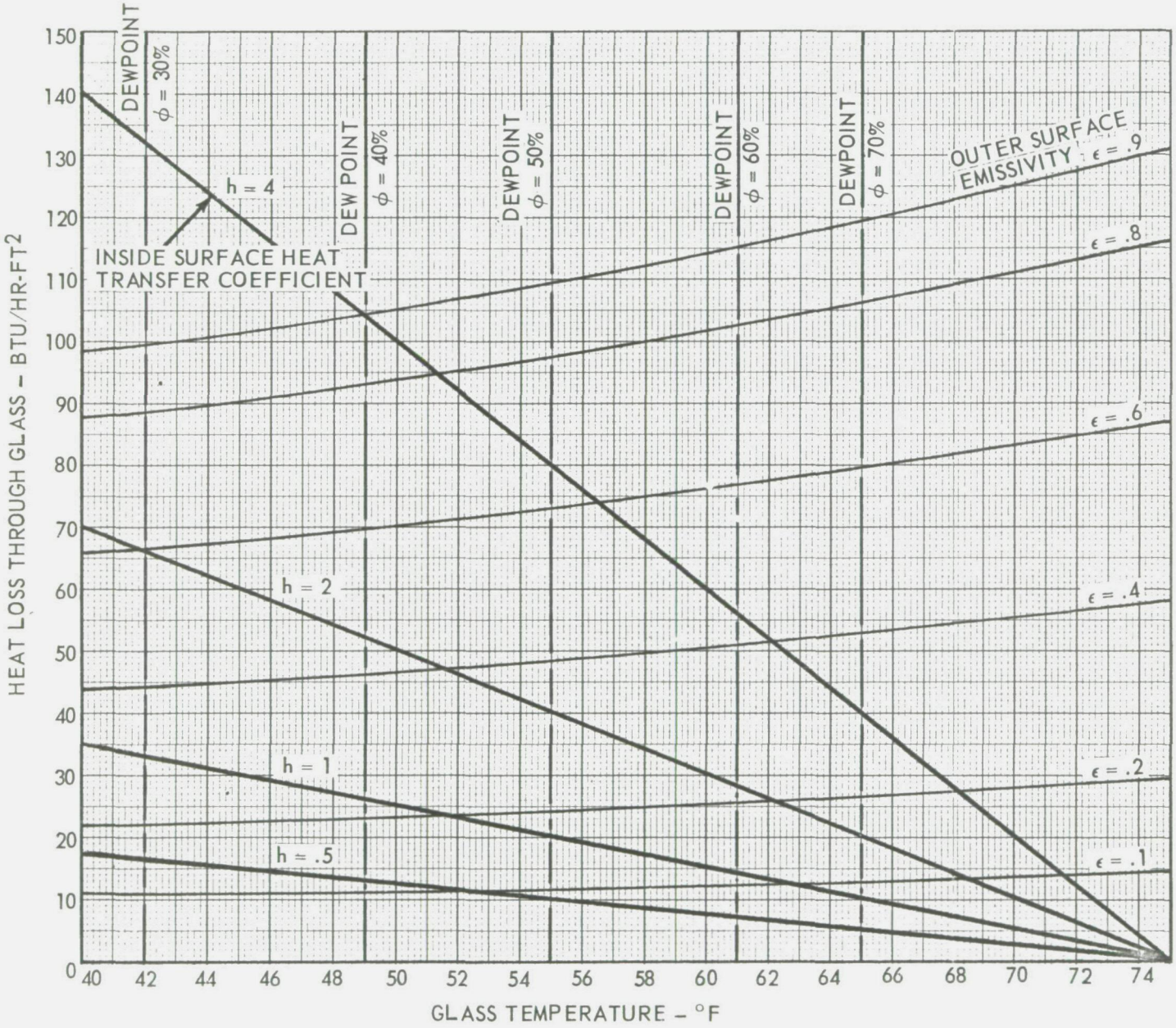


Figure 16-21 WINDOW HEAT LOSS FROM A SINGLE GLASS PANE – LUNAR COLD SIDE

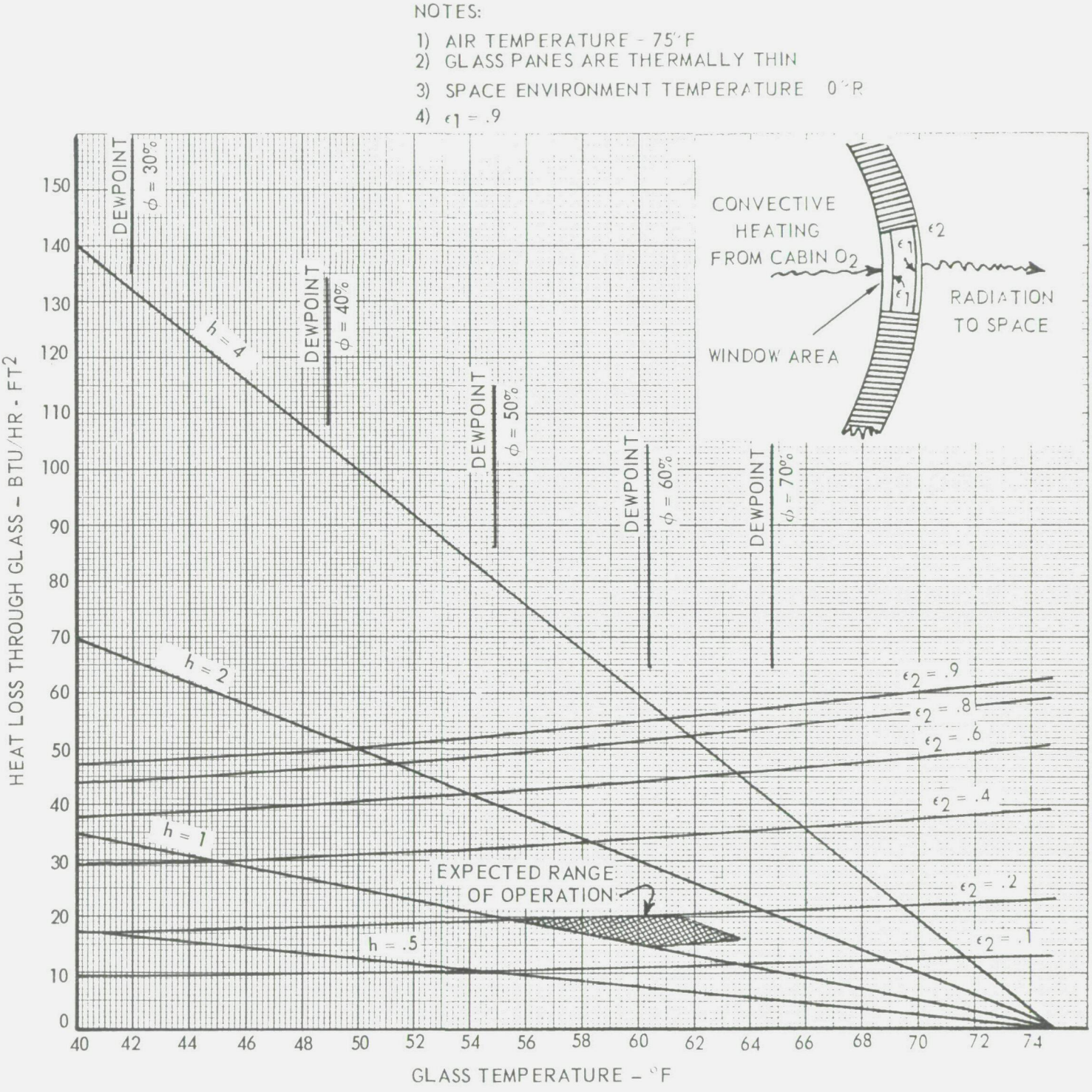


Figure 16–22 WINDOW HEAT LOSS FROM A DOUBLE GLASS PANE – LUNAR COLD SIDE

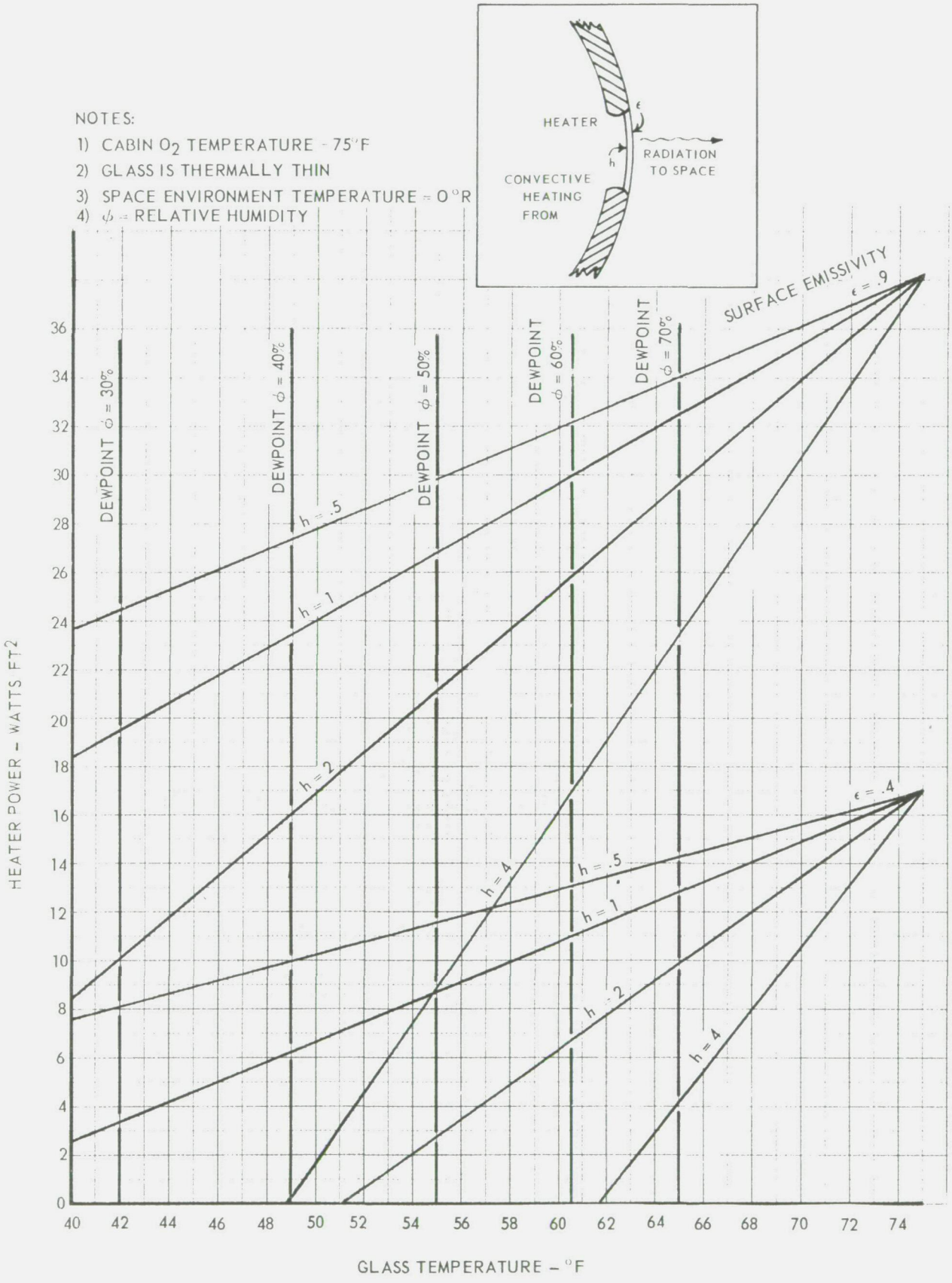


Figure 16–23 ELECTRICAL HEATER POWER REQUIRED FOR WINDOW DE-FOGGING

A decrease in the window emissivity to the required level without excessively impairing visual properties can be obtained with a thin metallic film. An example of one film which would be satisfactory is PE-81E film manufactured by the Libbey-Owens-Ford Glass Company. This film is a reflective gold coating. The approximate properties of this film are presented in Figure 16-24.

The low emissivity coating approach was selected for the LEM window because it provides a simple passive means to control the heat loss. This method is inherently reliable, lightweight, and requires no complicated controls.

When the vehicle is in the cold environment, the heat losses through the window have a significant effect on the ECS performance. Figure 16-25 shows the heat loss from the window as a function of window emissivity. For an emissivity of 0.2 and a heat transfer coefficient of 1.0 BTU/hr-Ft.^{2}-^{o}F., the heat loss is approximately 1100 BTU/hr. for the entire window area. This value is compatible with the capability of the ECS.

Heating of the vehicle when the window is oriented toward the sun causes a much more difficult problem. Since the maximum intensity of solar heating occurs within the visible wavelength range, a window that transmits visible light will also transmit a large amount of solar energy. The solar energy is absorbed within the vehicle and results in a large heat load which the ECS must dissipate. Of the solar energy striking the window, part of it is transmitted through the window, part is absorbed within the window, and part is reflected back to space. Figure 16-26 shows the total heat absorbed through the window as a function of the transmissivity and absorptivity of the glass. From Figure 16-26, it can be seen that for any glass transmissivity which provides adequate vision, the heat load on the ECS will be excessive.

During translunar flight and during periods of descent and rendezvous when all the window area is not required for visibility, the heat load can be greatly reduced by installing a reflective curtain inside the window. A curtain with a solar reflectivity of 0.9 will reduce the heat load to about 48 BTU/hr.-ft.^{2} or about 2200 BTU/hr. for the entire window area. This still imposes a significant load on the ECS but it occurs for a relatively short period of time.

During operation on the hot side of the lunar surface, the internal window curtain will not be adequate. Thermal control during this period requires that a low absorptivity shutter be installed on the exterior of the window. This type shutter can be incorporated for hot side landings and will reduce the heat absorbed through the windows to the acceptable level of approximately 230 BTU/hr.

16.3.4.4 Structural Heat Leakage

Heat leakage through the structure must be controlled to prevent excessive heat loads on the ECS. The heat leakage from the LEM cabin consists of: (1) heat transferred through the window area, (2) heat conducted through structural attachments such as the landing gear, and (3) heat

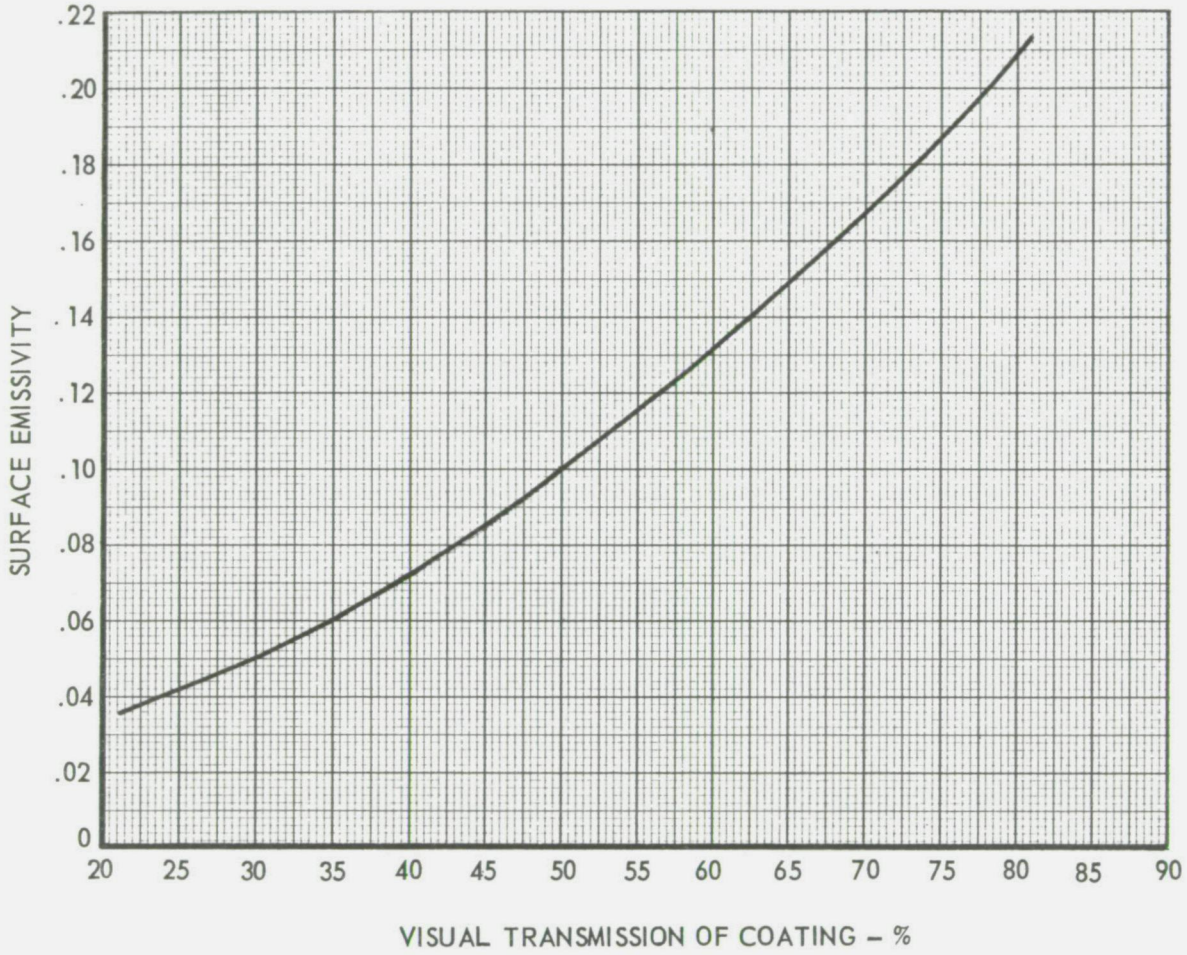
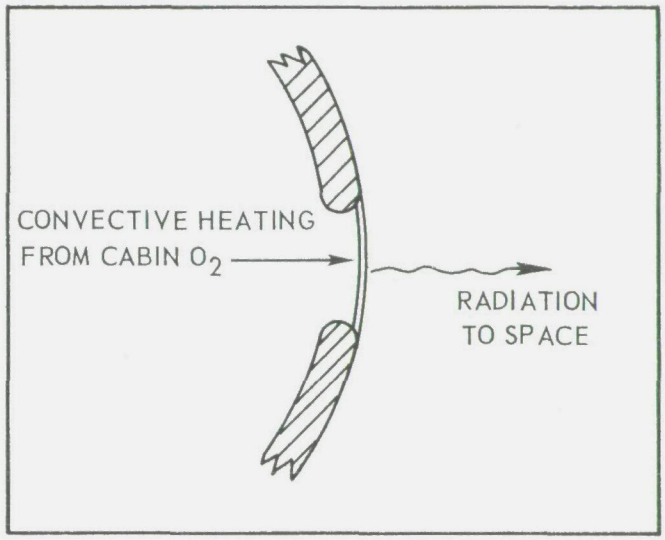


Figure 16–24 PROPERTIES OF COATING PE – 81E



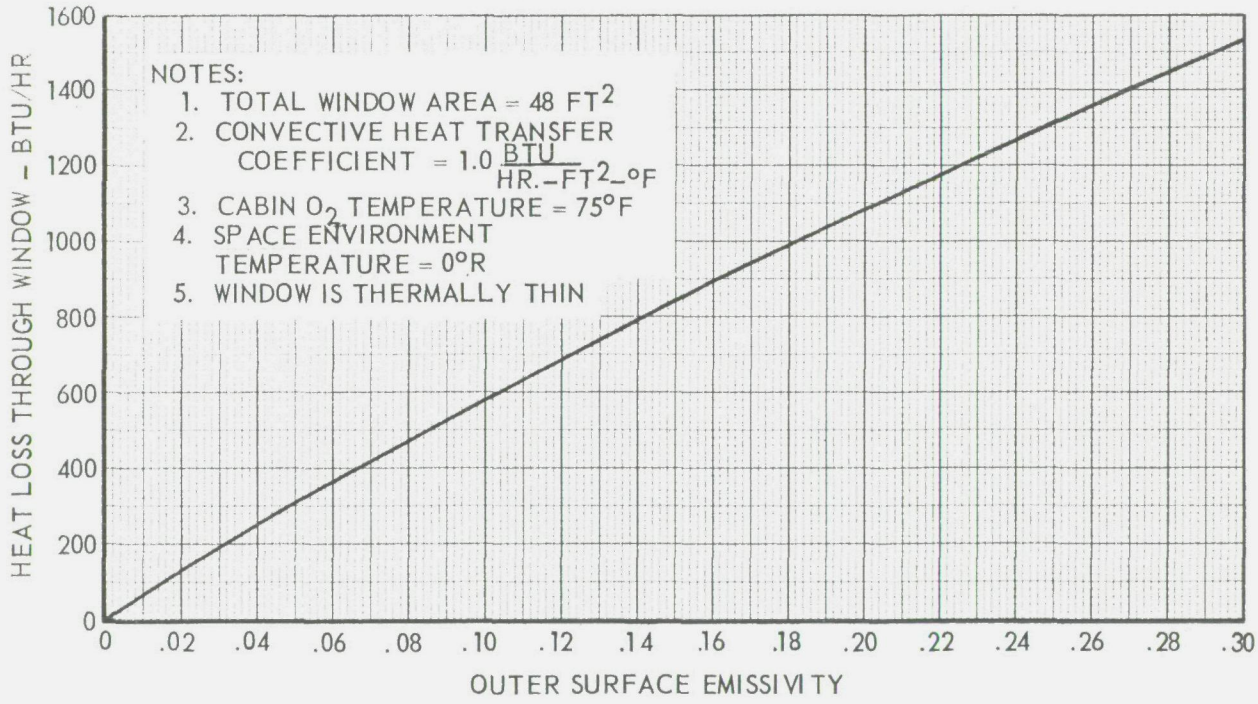


Figure 16–25 EFFECT OF OUTER SURFACE EMISSIVITY ON MAXIMUM WINDOW HEAT LOSS

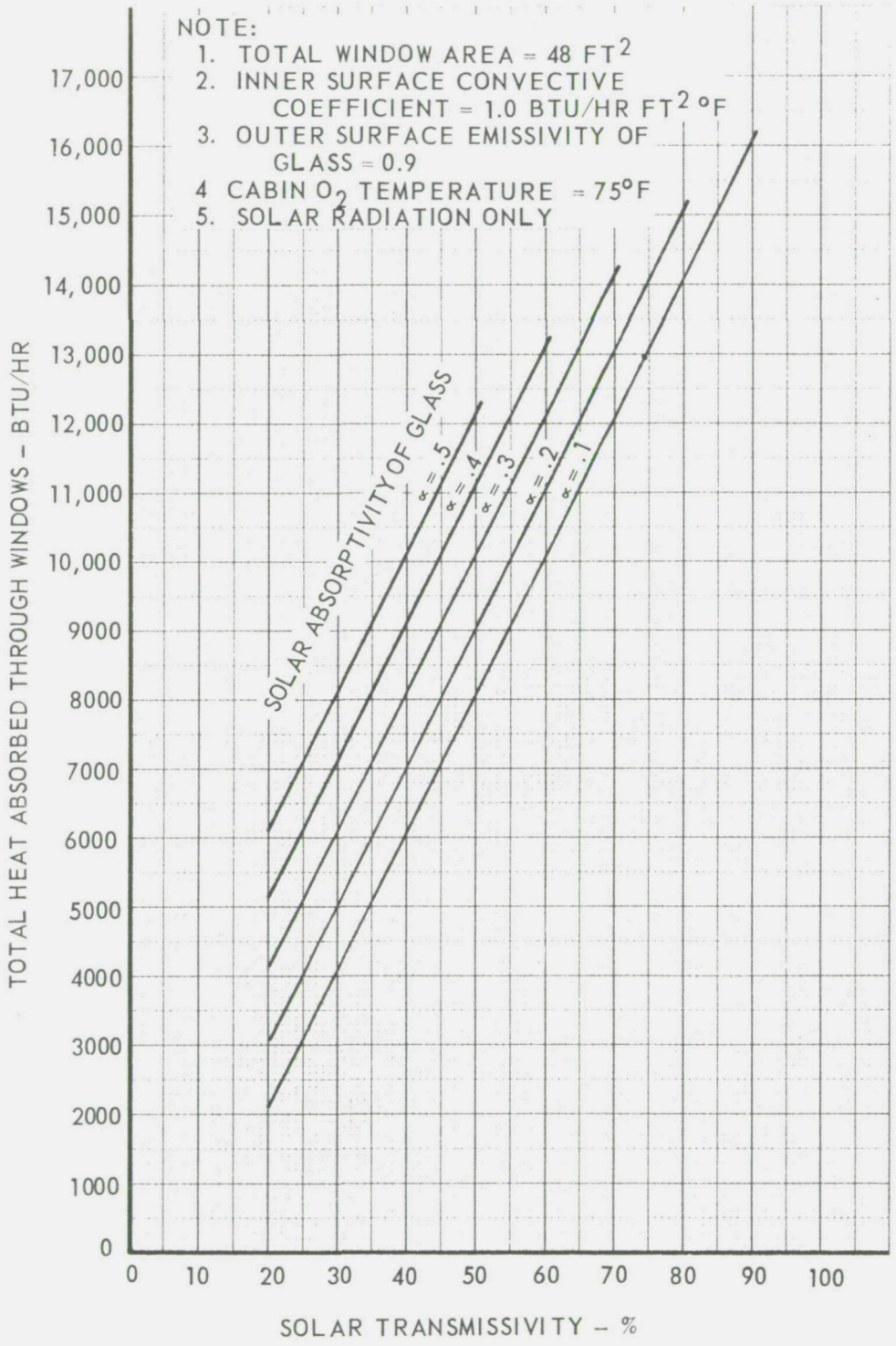


Figure 16-26 HEAT TRANSFERRED THROUGH WINDOWS AS A FUNCTION OF WINDOW PROPERTIES

transferred through the cabin walls. The effect of these heat loads on the ECS depends on whether the heat is transferred from the cabin (cold environment) or whether the heat is transferred into the cabin (hot environment). Small heat losses from the cabin do not present a significant problem since excess heat is generated by electrical equipment and by the crew. If the rate of heat loss exceeds the rate of heat generated, a heating system must be provided to maintain an acceptable temperature. Heat transferred into the cabin from external sources requires an increased environmental system cooling capacity and larger space radiators. During lunar orbits and lunar hot side landings, the increased heat load requires additional quantities of expendable coolant. This adds significantly to the weight of the environmental control system.

Accurate analysis of heat transfer through the structural attachments is difficult until the actual vehicle has been designed. These attachments can be thermally isolated and the surfaces treated to obtain a low emissivity so that the heat losses, or heat absorbed by the cabin, are kept to a minimum. Based on representative designs, the structural heat loss in the cold environment may be kept to approximately 400 BTU/hr. with good design procedures. This number is compatible with the ECS capabilities. Maximum heating through the structural attachments in the hot environment is estimated to be approximately 100 BTU/hr.

Because of the large surface area of the cabin walls, special consideration must be given to their thermal design. During operation in the cold environment, which will be experienced during landings on the cold side of the lunar surface and for short periods of time in lunar orbit, the heat loss to space becomes excessive for an unprotected structure. This heat loss to space can be reduced by the application of low emissivity surface coatings and/or by the addition of insulation to the structure. The low emissivities are derived from polished metal surfaces such as aluminum or gold. Highly polished surfaces are difficult to maintain, and a practical minimum emissivity for aluminum is approximately 0.1. With gold coatings, emissivities slightly lower than 0.1 can be obtained.

Figure 16-27 shows the heat loss to space by the LEM walls as a function of emissivity. A wall surface area of 220 ft.^{2} was used in the calculations. The figure indicates that the structural heat loss for an emissivity of 0.1 is approximately 3000 BTU/hr. Heat losses by the structural attachments and through the window area must be added to this value to determine the total vehicle heat loss. During lunar stay, the heat dissipated inside the cabin can drop to a value as low as 2200 BTU/hr. which means that either additional thermal protection for the walls is required or a cabin heater must be included in the ECS.

Additional wall thermal protection can be provided by insulation. Since the LEM is not required to withstand aerodynamic pressures and heating loads, insulation may be applied to the external surface. Externally applied insulations provide a lower thermal conductivity due to the high vacuum of space. Lightweight multi-layer radiation shield insulation, such as Linde SI-4 or National Research Corporation NRC-2, provides considerable heat blockage with small thicknesses and weight. Figure 16-28 shows

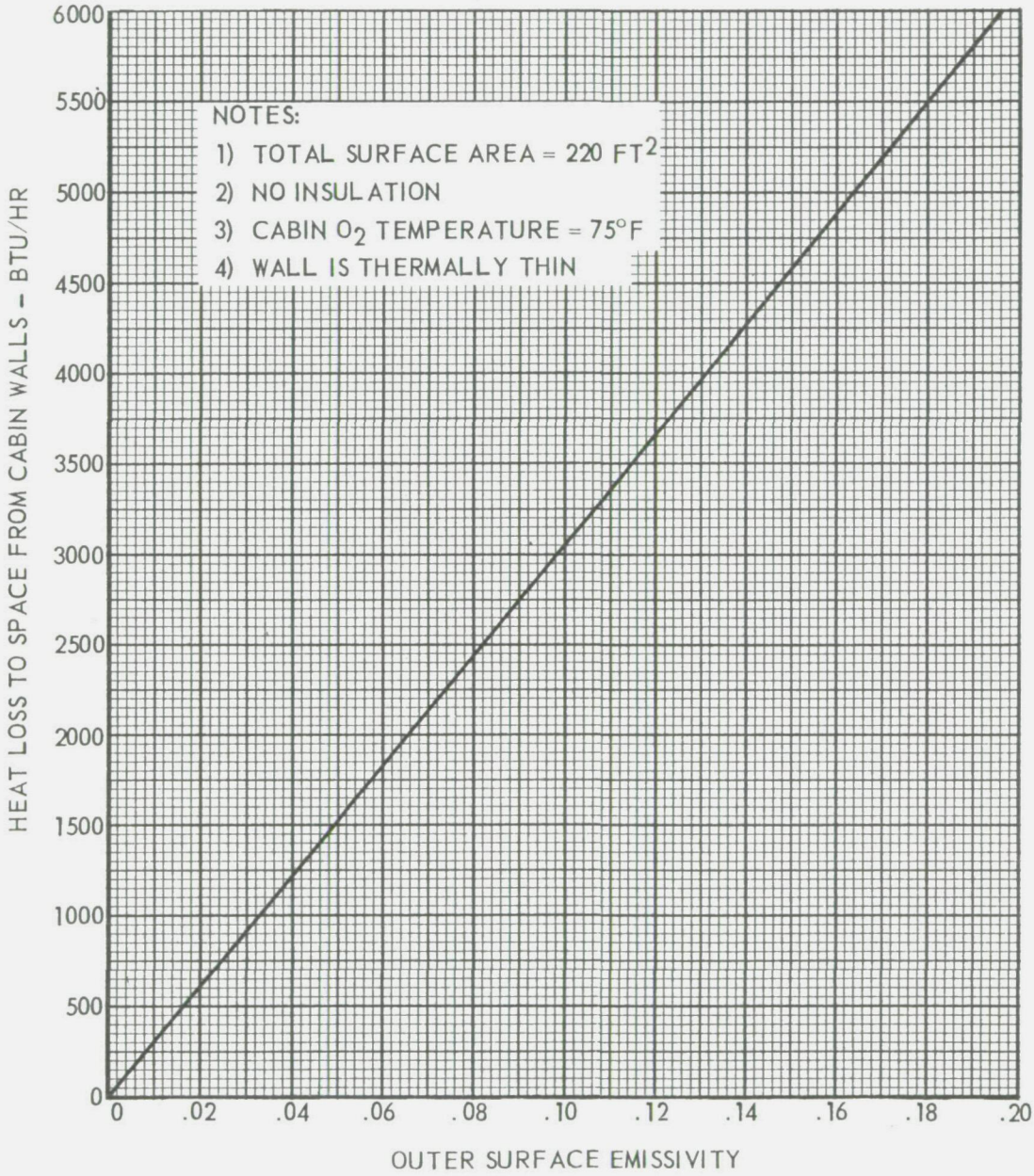


Figure 16-27 EFFECT OF VEHICLE SURFACE EMISSIVITY ON HEAT LOSS FROM CABIN WALLS (LUNAR COLD SIDE OPERATION)

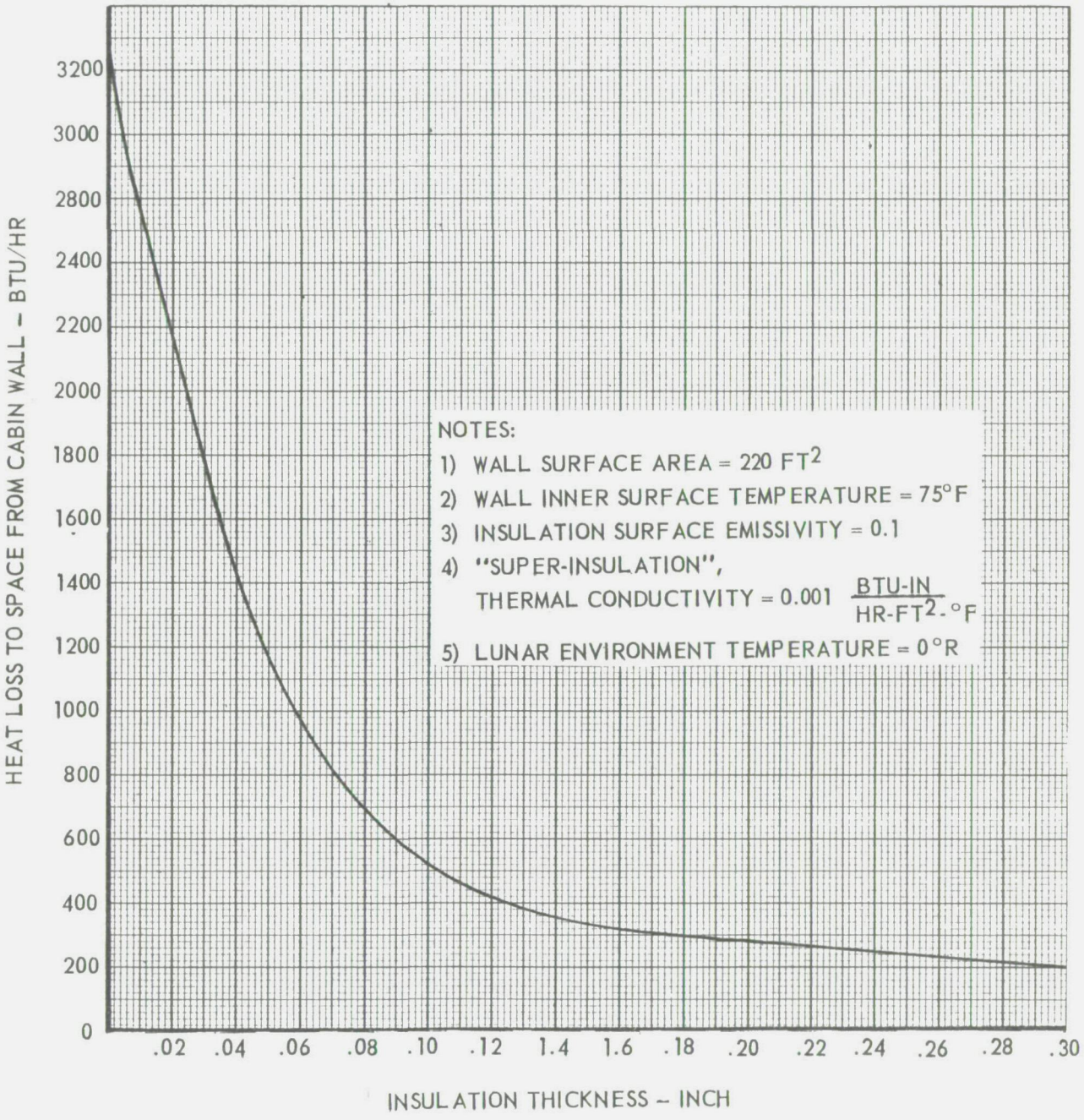


Figure 16–28 EFFECT OF INSULATION THICKNESS ON HEAT LOSS FROM CABIN WALLS (LUNAR COLD SIDE OPERATION)

the effect of thickness of a typical super-insulation on the heat loss to space through the cabin wall. An outer surface emissivity of 0.1 is used. This is representative of bright aluminum foil or aluminized mylar. Outer layers of thin aluminum foil or mylar film may cause a handling problem during preflight maintenance and checkout. Additional studies are required to define the specific means for installation and the handling protection required. A minimum practical super insulation thickness is approximately 0.1 inch. With this effective thickness, Figure 16-28 indicates that approximately 450 BTU/hr. will be lost to space through the cabin walls. This value is compatible with the ECS capabilities without special heating provisions.

When the LEM lands on the hot side of the moon, the entire vehicle surface will be heated by the sun or by the hot lunar surface. An estimate of the radiant heat input to the spacecraft can be made by considering 1/4 of the surface area (which is the projected area of a sphere) heated by the sun and the rest heated by the lunar surface. The solar heating depends on the solar absorptivity and the emissivity of the vehicle surface. The heating rate for no wall insulation is shown in Figure 16-29 as a function of surface emissivity and solar absorptivity. This figure shows that the heat absorbed is affected considerably by surface emissivity and moderately by solar absorptivity. This result is obtained because the majority of the vehicle surface is heated by lunar radiation. The absorptivity of the vehicle surface to lunar radiation is approximately equal to the emissivity. A polished surface such as aluminum foil has an emissivity of approximately 0.1 and a solar absorptivity of approximately 0.15. Figure 16-29 indicates that with this surface the spacecraft will absorb approximately 4,400 BTU/hr., which is a very high heat load. Figure 16-30 shows the heat absorbed by the walls with various thicknesses of "super-insulation" added. This figure shows that the heat loads are reduced to acceptable levels with small thicknesses of super-insulation.

In order to determine the optimum insulation thickness for the cabin walls, a trade study was required. Evaporation of cooling water is used during lunar hot side landings to augment the radiator since the radiator cannot reject the total heat load at the required low temperature of 40°F. The total weight penalty is the sum of the insulation and water (including associated attachments and tankage) weights. Since insulation weight is proportional to thickness and cooling water weight is inversely proportional to thickness, the minimum total weight penalty occurs when the insulation weight is equal to the cooling water weight. The total weight penalty for an optimum insulation-cooling water arrangement is shown in Figure 16-31 as a function of lunar hot side stay time. The optimum insulation thickness and total weight penalty for an insulation thickness of 0.10 inches are also shown. From Figure 16-31 it may be seen that the weight for an insulation thickness of 0.1 inches does not vary significantly from the optimum weight for missions of 24 to 160 hours duration. For a lunar stay time of 160 hours, the weight penalty with 0.10 inches of insulation is approximately 12 pounds.

As indicated by the previous discussion, the heat leak through the cabin walls cannot be adequately controlled by coatings alone for either hot side or cold side lunar landings. Because of the extremely low vacuum thermal conductivity of the insulation, a minimum thickness is sufficient.

NOTES:

1) TOTAL WALL AREA = 220 FT^2
2) NO INSULATION
3) LUNAR SURFACE TEMPERATURE = 250°F

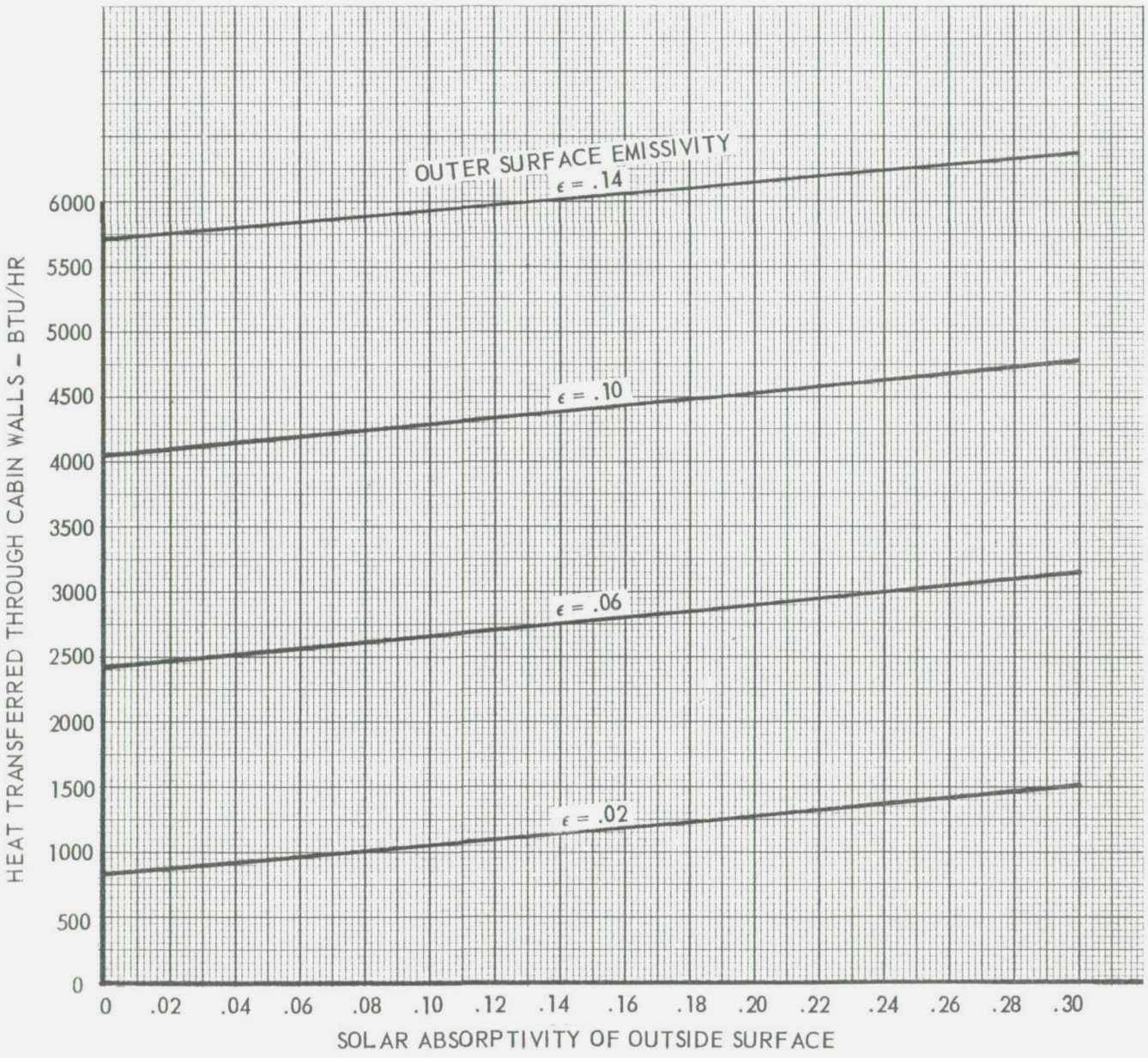


Figure 16-29 EFFECT OF VEHICLE SURFACE PROPERTIES ON HEAT TRANSFERED THROUGH THE CABIN WALLS (LUNAR HOT SIDE)

NOTES:

1) TOTAL WALL SURFACE AREA = 220 FT^2
2) SUPER INSULATION,
   THERMAL CONDUCTIVITY = 0.001 $\frac{BTU\text{-}IN}{HR\text{-}FT^2 \cdot °F}$
3) INSULATION SURFACE EMISSIVITY = 0.1
4) INSULATION SURFACE SOLAR ABSORPTIVITY = 0.15

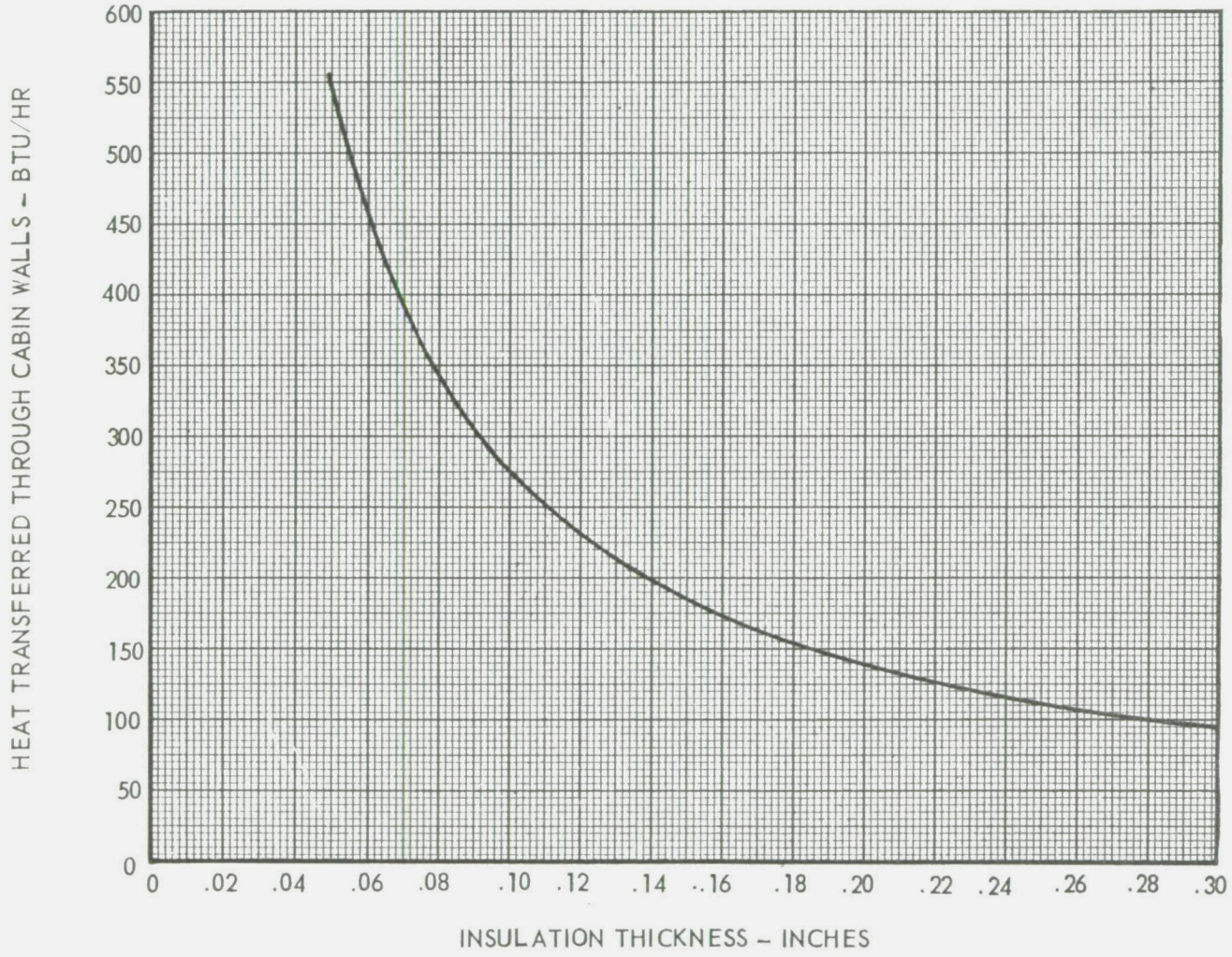


Figure 16–30 EFFECT OF INSULATION THICKNESS ON HEAT TRANSFERRED THROUGH THE CABIN WALLS
(LUNAR HOT SIDE)

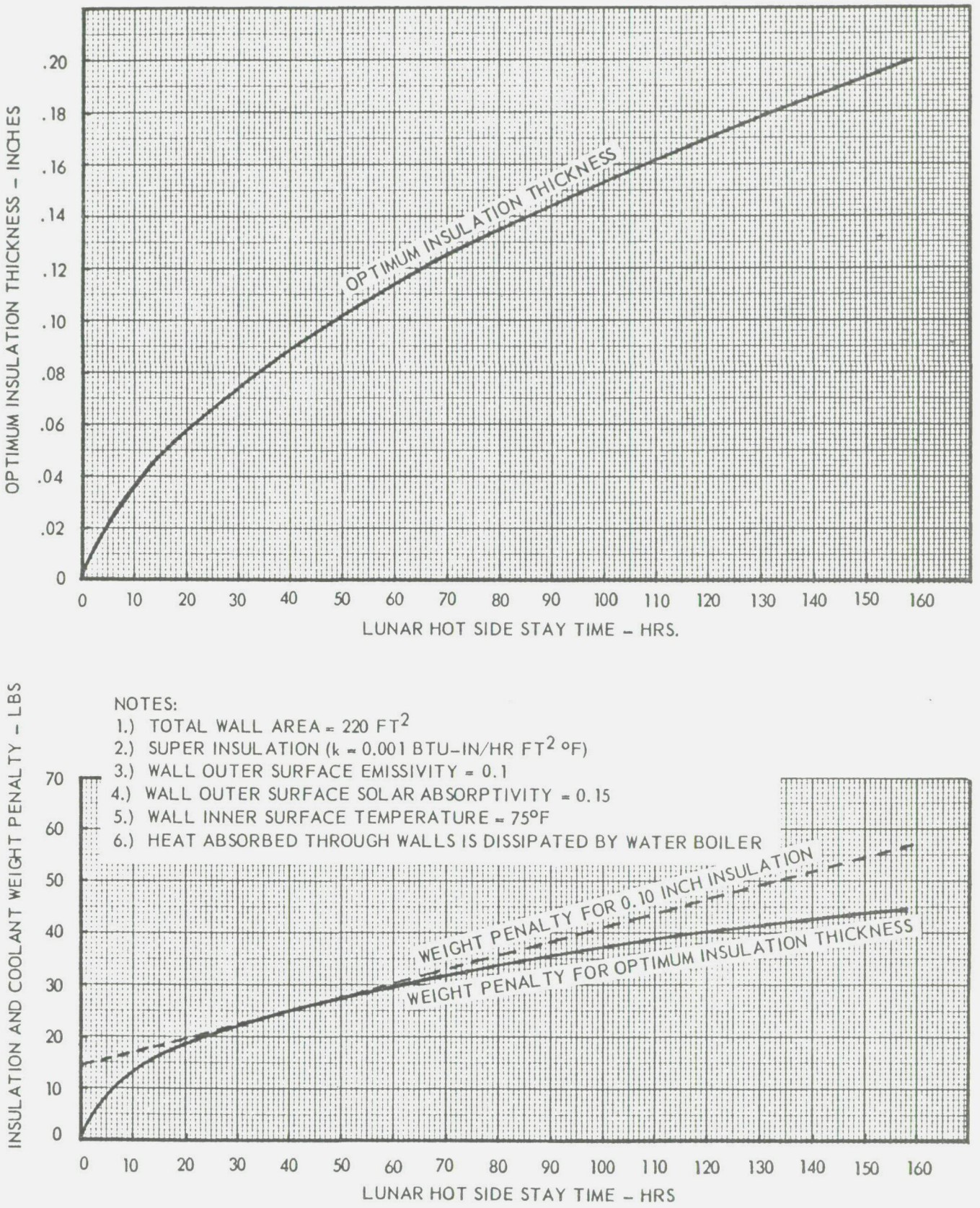


Figure 16–31 WEIGHT PENALTY FOR CABIN WALL THERMAL PROTECTION

(LUNAR HOT SIDE)

Therefore, the cabin walls will be insulated by an external layer of 0.10 inches of "super-insulation" with a low emissivity outer surface. This low emissivity surface may be aluminum foil, aluminized mylar, or a gold film.

A summary of the heat leakage through the window, structural attachments and cabin walls of the recommended vehicle is shown in Table 16-XI.

16.3.4.5 Thermal Control Loop Analysis

A detailed analysis of the thermal control loop for the selected vehicle and mission was made to obtain performance data and to verify assumptions made during the initial trade study. This section presents the data obtained from this analysis.

16.3.4.5.1 Thermal Loads

The internal thermal loads include the following four classes:

(a) Crew Metabolic - The thermal loading attributed to the crew (latent and sensible) varies from approximately 350 BTU/man-hr. at rest to 1000 BTU/man-hr. during periods of strenuous activity. An average loading of 500 BTU/man-hr. can be assumed for design purposes if provisions are made for perturbations due to varying crew tasks. Metabolic heat increases the temperature of the environmental oxygen and must be removed at a temperature level compatible with crew comfort (68 - 76°F.). In addition, a lower limiting condition is established by the temperature at which excess water vapor can be removed from the oxygen. This temperature is determined by the specified relative humidity and is approximately 40°F. for the LEM.

(b) Structural Leakage - The magnitude of the thermal loading attributed to structural leakage is determined by the external environment, attitude of the vehicle, vehicle surface properties, and structural heat paths to the vehicle interior. Evaluation of these criteria in Paragraph 16.3.4.4 shows that the design boundary conditions occur during low orbit over the sub-solar point, rest at the sub-solar point, and rest at lunar midnight. The magnitude of the heat leakage during the above limiting conditions is approximately 2300 BTU/hr., 600 BTU/hr. and -2000 BTU/hr. respectively. Since this heat load is initially transferred to the environmental oxygen, the temperature level of removal is identical to that for crew metabolic; namely, 40°F. to 76°F.

(c) Electrical and Electronic Equipment - All electrical and electronic equipment operates with losses which contribute to the thermal load within the vehicle. These energy losses appear as heat which must be removed at or near the design operating temperature of the equipment in order to avoid degradation and reduced reliability. The mission equipment thermal load profile is shown in Figure 16-32. The temperatures at which these loads must be removed varies between 85 and 160°F.

TABLE 16 – XI SUMMARY OF CABIN HEAT LOADS

ITEM	LUNAR ORBIT, LANDING AND RENDEZVOUS		LUNAR SURFACE OPERATIONS	
	MAX LOAD BTU/HR	MIN LOAD BTU/HR	COLD SIDE	HOT SIDE
WINDOW	2200	-1100	-1100	+230
STRUCTURAL ATTACHMENTS	+100	-400	-400	+100
CABIN WALL	0	-500	-500	+270
TOTAL	2300	-2000	-2000	+600

NOTE:
MINUS SIGN DENOTES LEAKAGE FROM CABIN
PLUS SIGN DENOTES LEAKAGE TO CABIN

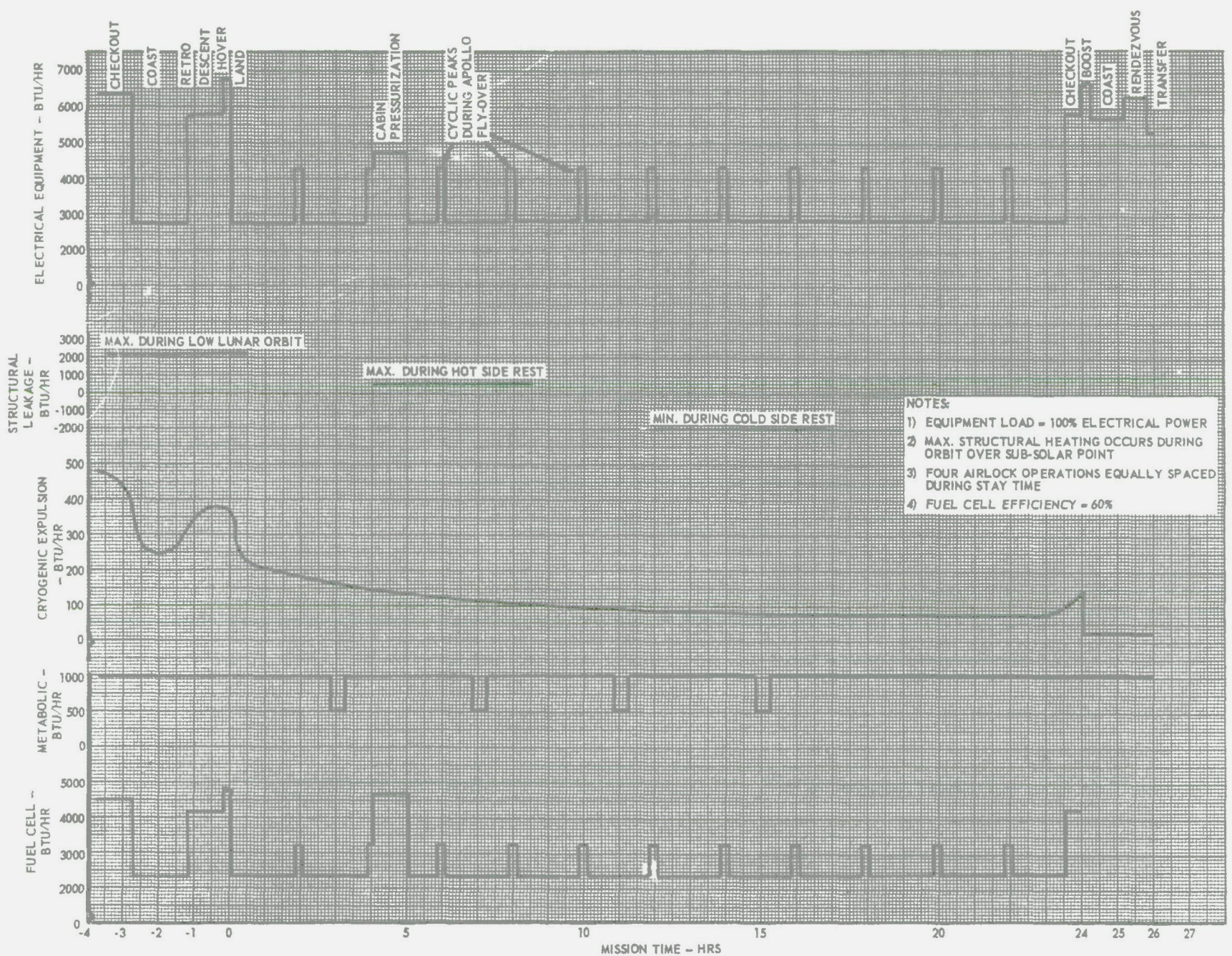


Figure 16-32 VEHICLE THERMAL LOAD SUMMARY

(d) Electrical Power System - The electrical power system operates with efficiency losses which appear as waste heat. The waste heat for the LEM fuel cell is approximately 60% of the total electrical load. In addition, the electrical power system cryogenic fuel supply tanks, as well as the ECS oxygen supply tanks, require heat inputs for fluid pressurization. The cryogenic tank pressurization heat input requirements are also shown in Figure 16-32. The temperature level of waste heat removal from the fuel cell is 141°F. to 500°F. The temperature level of heat addition to the cryogenic tanks must be sufficiently above the fluid temperature to ensure sufficient heat transfer rates with low volume heat exchangers.

The mission profile for each of the above heat sources and sinks comprising the total vehicle thermal loading is included in the summary curve of Figure 16-32.

16.3.4.5.2 Heat Transport Loop

A schematic of the selected thermal control loop is shown in Figure 16-33. The component arrangement shown is dictated primarily by the necessity of routing the glycol flow through the following successively higher temperature sources and sinks.

(a) Suit circuit heat exchanger

(b) Cabin heat exchanger

(c) Guidance package

(d) Electrical equipment

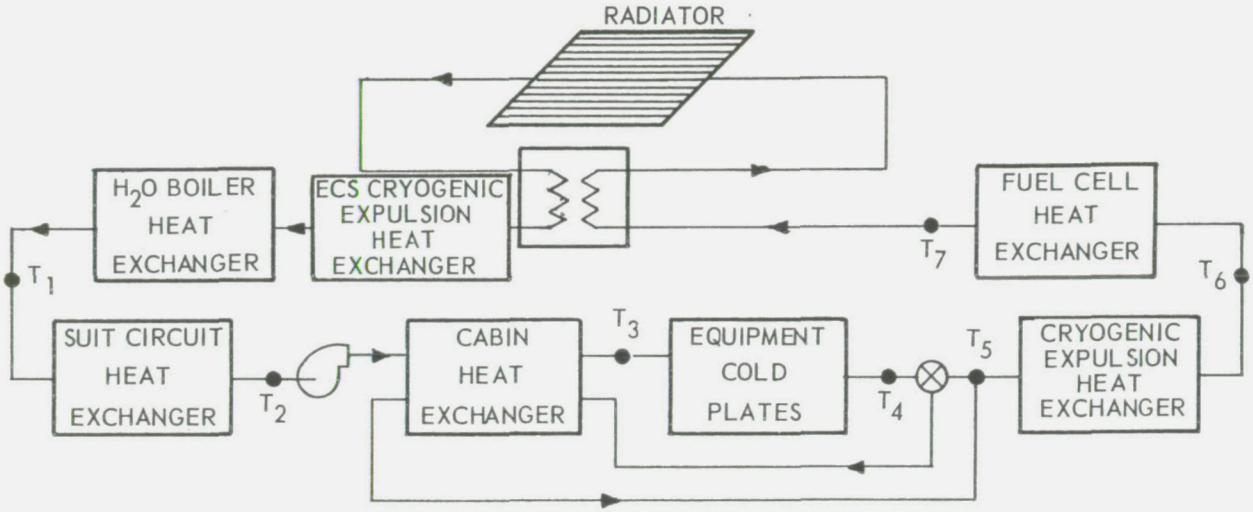
(e) Fuel cell cryogenic expulsion heat exchangers

(f) Fuel cell heat exchanger

The suit circuit heat exchanger reduces the oxygen temperature to approximately 45°F. in order to condense water in the air stream and allow removal for humidity control. A heat load of approximately 1000 BTU/hr. is removed from the oxygen by this heat exchanger. This allows sufficient capacity for removal of the crew metabolic heating during operation in the pressure suits with an unpressurized cabin.

The glycol pumps are located immediately downstream of the suit circuit heat exchanger. Their inclusion at this point allows introduction of pump heat loads downstream of the low temperature suit circuit loads and enhances pump reliability by placing them in the low temperature side of the loop. Positive displacement pumps are used in order to provide constant glycol flow over a wide range of temperatures and pressures.

Downstream of the pumps, sufficient glycol is routed through the cabin heat exchanger to control the cabin temperature to the desired 68-76°F. range. From the cabin heat exchanger the glycol flows to the various equipment cold-plates in order of ascending temperature requirements. The



MISSION PHASE	HEAT LOAD – BTU/HR.					
	SUIT CIRCUIT HEAT EXCHANGER	CABIN CIRCUIT HEAT EXCHANGER	ELECTRICAL EQUIPMENT COLD PLATES	CRYOGENIC EXPULSION HEAT EXCHANGER	FUEL CELL HEAT EXCHANGERS	TOTAL
DESCENT	1,000	3,180	5,900	–375	4,800 *	14,600
COLD SIDE REST	1,000	–1,300	3,600 *	–75	3,240	6,465
HOT SIDE REST	1,000	1,300 *	3,600	–75	3,240	9,240

MISSION PHASE	GLYCOL FLOW RATE LBS/HR.
DESCENT	127
COLD SIDE REST	52
HOT SIDE REST	92

MISSION PHASE	MAXIMUM GLYCOL TEMPERATURE – °F						
	T_1	T_2	T_3	T_4	T_5	T_6	T_7
DESCENT	40	51	86	146	146	141	191
COLD SIDE REST	40	66	68	160	127	125	207
HOT SIDE REST	40	55	75	127	127	126	172

* DENOTES COMPONENT DETERMINING GLYCOL FLOW RATE

Figure 16–33 HEAT TRANSPORT LOOP PERFORMANCE SUMMARY

glycol outlet temperature from the final cold plate does not exceed 160°F. Provisions are included to route part of the hot glycol from the last equipment cold plate back through heating tubes in the cabin heat exchanger. This heating function is required only during low power loadings on the cold side of the lunar surface. From the equipment, the glycol loop passes through the fuel cell cryogenic expulsion heat exchangers and then to the fuel cell cooling heat exchanger.

The required glycol flow rate is determined from (1) the magnitude of the loads and (2) the total allowable temperature increase between the suit circuit heat exchanger and the downstream side of the last equipment cold plate. This holds true provided the specified temperature requirements of the individual loads are satisfied. Satisfaction of these individual heat loads results in off-design glycol flow rates except at the peak loading conditions. This problem can be minimized by using different flow rates for lunar orbit and landing, and lunar rest. The vehicle heat loads determining glycol flow rates are shown in Figure 16-33, for the boundary conditions of lunar orbit, hot side lunar rest, and cold side lunar rest. Figure 16-33 also shows flow rates and critical glycol temperatures for the above boundary conditions.

16.3.4.5.3 Radiator Analysis

Analysis to determine the radiator area was based on the following assumptions:

(a) Plane area radiating surface facing directly away from the lunar surface.

(b) Negligible incident radiation from lunar terrain (i.e., flat lunar surface).

(c) Negligible "dusting" of the radiating surface.

The radiator must be capable of dissipating the maximum vehicle heat loads without freezing under minimum loading conditions. The thermal capacitance of the loop will partially dampen out the ten minute cyclic heat load peaks which occur during the Command Module fly-over (Figure 16-32) allowing use of a radiator design point based on the time weighted average loads. However, the glycol flow rate must be based on the maximum rather than the average loading to prevent local component overheating during these peaks. This results in slight deterioration of radiator performance due to off-design operation, but is preferable to the reliability penalty associated with variable flow rate glycol pumps.

For the special case where the radiator equilibrium temperature (T_e) is approximately 0°R. (i.e., lunar cold side), the required radiator surface area is determined by:

(a) The temperature level of heat rejection.

(b) Emissive properties of the radiator coating.

(c) Radiator fin effectiveness.

For the time weighted average vehicle heat load and maximum heat load glycol flow rate, the radiator inlet temperature is approximately 140°F. With the inlet temperature to the suit circuit heat exchanger specified as 40°F., the required temperature drop across the radiator is 100°F. A titanium dioxide coated radiator has an emissivity of approximately 0.83 at this average radiator temperature. The radiator fin effectiveness factor indicates the thermal efficiency of the radiator as a function of the temperature drop between the fluid in the radiator tube and a point on the radiator fin midway between tubes. Generally, fin effectiveness increases as tube spacing decreases. A decrease in tube spacing, however, results in increased radiator weight per unit area. Reference 13 shows that a fin effectiveness factor of approximately 0.92 can be achieved with redundant radiator tubes before prohibitive unit weight is encountered.

With these radiator characteristics defined, the required area can be expressed as:

$$A_R = \frac{\dot{\omega}\, Cp \left[\left(\frac{T_i}{T_o} \right)^3 - 1 \right]}{3\sigma \epsilon_R \eta_f T_i^3}$$

$\dot{\omega}$ = glycol flow rate - $\frac{lb}{hr}$

Cp = glycol specific heat - $\frac{BTU}{lb\ ^oF.}$

T_i = radiator inlet temperature - ^oF

T_o = radiator outlet temperature - ^oF

σ = Stefan-Boltzmann constant $\frac{BTU}{HR\ FT^2\ {}^oR^4}$

ϵ_R = Surface emissivity

η_f = Fin effectiveness factor

Solution of the above equation yields a required radiator area of 35 ft.^{2} for the LEM when operating on the dark side of the moon. This area is based on the lunar rest, time weighted average total vehicle heat load.

Maintenance of both the vehicle thermal balance and the 40°F. glycol inlet temperature to the suit circuit heat exchanger during off-design radiator operation requires a reduction in the radiator heat rejection rate. This reduction can be obtained by (1) use of a fluid by-pass to route part of the fluid flow around the radiator and (2) use of a regenerative heat exchanger.

Figure 16-34 shows the range of vehicle heat load controllable by both of these methods prior to failure due to radiator freezing or "slushing". The regenerative heat exchanger allows a 17% greater load variation than does the by-pass arrangement. Therefore, the regenerative heat exchanger was selected for radiator thermal control.

As shown in Paragraph 16.3.2.5, a water boiler is required to augment the radiator during peak heat load conditions. Apportionment of the heat loads between the radiator and the water boiler during lunar hot side operation requires an analysis of over-all effective vehicle weight and volume. Effective vehicle weight is defined here to differentiate between items which are: (1) fixed weight and boosted back into lunar orbit, (2) fixed weight, but off-loaded on the lunar surface, and (3) expendable weight.

For the time weighted average vehicle heat load and the glycol flow rate for maximum loading, the radiator inlet temperature on the lunar hot side is approximately 140°F. Under this condition, 7000 BTU/Hr. must be rejected by the space radiator and/or water boiler to effect the required 100°F. temperature drop resulting in a 40° F. glycol temperature at the inlet to the suit circuit heat exchanger. Optimization of the apportionment of the heat rejection function is based on an effective vehicle weight penalty of 0.9 lb/ft^2, 2.25 lb/ft^2, and 1.125 lbs. of H_2O/1000 BTU for off-loaded radiators, fixed radiators and water boilers respectively (Reference 15). The glycol temperature at the radiator outlet (T_o) actually determines the magnitude of the contribution of the radiator and the water boiler toward dissipation of the total load. Therefore, this temperature is a variable in the optimization study.

As previously noted, the selected radiator has a surface emissivity of approximately 0.83 and a fin effectiveness factor of approximately 0.92. The variation of radiator equilibrium temperature (T_e) during hot side lunar rest and for times centered around the sub-solar point is shown in Figure 16-35.

The radiator area may be expressed as:

$$A_R = \frac{C_p \dot{\omega}}{2\sigma \epsilon_R \eta_F T_e^3} \left[\tan^{-1} \frac{T_o}{T_e} - \tan^{-1} \frac{T_i}{T_e} + 1/2 \ln \frac{\frac{T_e + T_o}{T_e - T_o}}{\frac{T_e + T_i}{T_e - T_i}} \right]$$

when $\dot{\omega}$ = glycol flow rate - lb/hr.

C_p = glycol specific heat - BTU/lb°F.

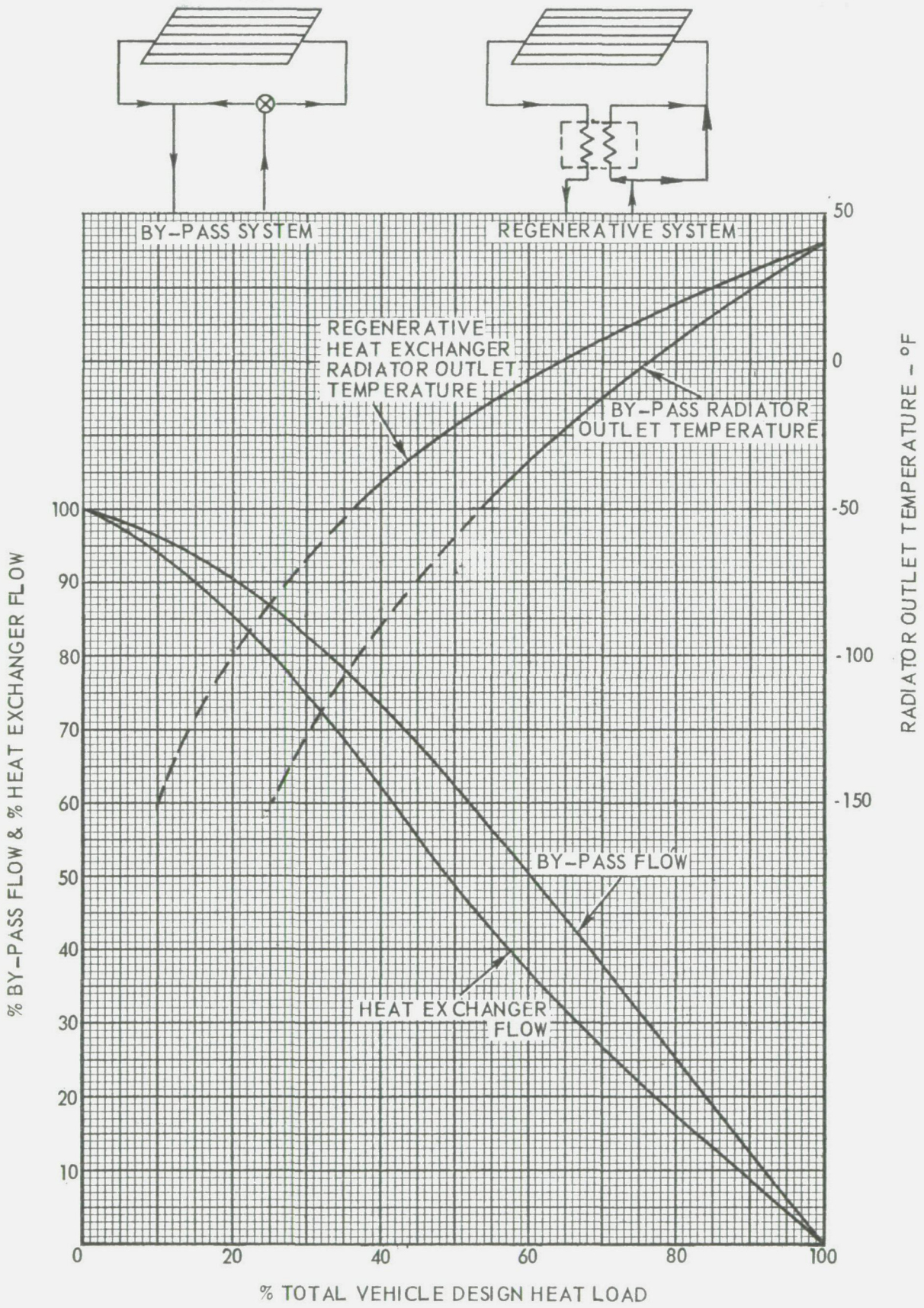


Figure 16-34 RADIATOR CONTROL PERFORMANCE WITH FLUID BY-PASS AND REGENERATIVE HEAT EXCHANGER

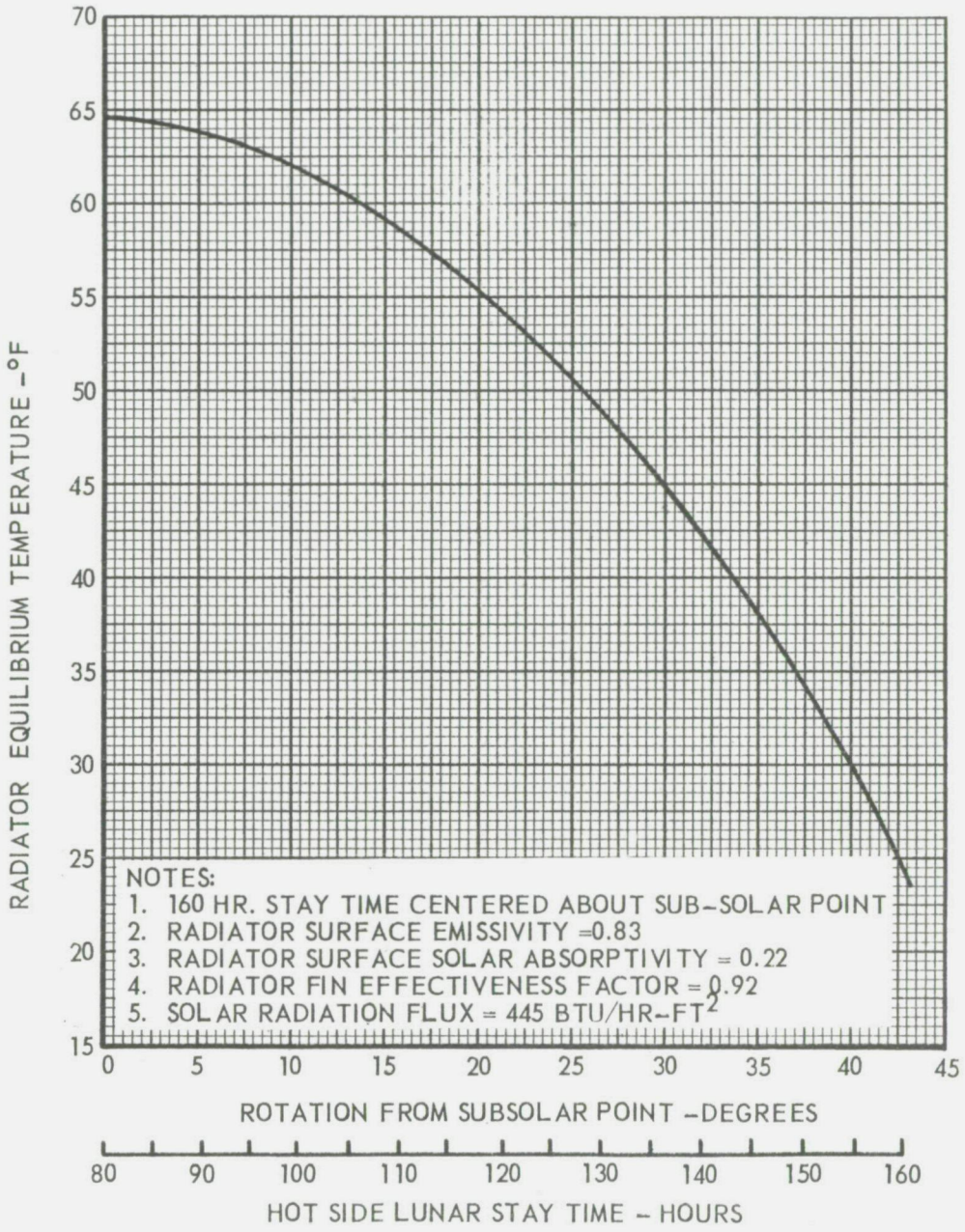


Figure 16-35 RADIATOR EQUILIBRIUM TEMPERATURE VARIATION (LUNAR HOT SIDE)

T_o = radiator outlet temperature

T_i = radiator inlet temperature

T_e = radiator equilibrium temperature

σ = Stefan-Boltzmann constant

ϵ_R = radiator surface emissivity

η_F = radiator fin effectiveness

The above equation and data can be solved to obtain radiator outlet temperatures as a function of radiator area and lunar surface stay time. Figure 16-36 is a plot of these data. From Figure 16-36, the radiator area and, therefore, the radiator weight penalty, for different outlet temperatures and lunar stay times can be determined.

The weight penalty for water boiler augmentation to further reduce the glycol temperature to 40^oF can be expressed as:

$$W_w = 1.125 \frac{WC_p}{h_{f_g}} (T_o - 40^oF)$$

where:

W_w = weight of water and tankage - lb/hr

W = glycol flow rate - lb/hr

C_p = glycol specific heat - BTU/lb^oF

h_{f_g} = H_2O heat of vaporization - BTU/lb

Figure 16-37 shows the required water boiler augmentation rate for different radiator areas and mission times. The effective weights for various system combinations are shown in Figure 16-38.

The value of radiator outlet temperature (T_o) resulting in the least effective vehicle weight for the combined systems determines the optimum radiator area for each given mission. Figure 16-39 shows the optimum radiator area for both fixed and off-loaded radiators as a function of lunar stay time (centered around the sub-solar point). This area is 53 ft.^{2} and 230 ft.^{2} respectively for the 24 hr. and 160 hr. lunar hot side missions with off-loaded radiators.

16.3.4.6 Analog Analysis of Atmosphere Control System

An analog computer was employed in the analysis of the selected atmosphere control system. The analog was employed to permit evaluation of the selected atmosphere control components based on over-all system performance.

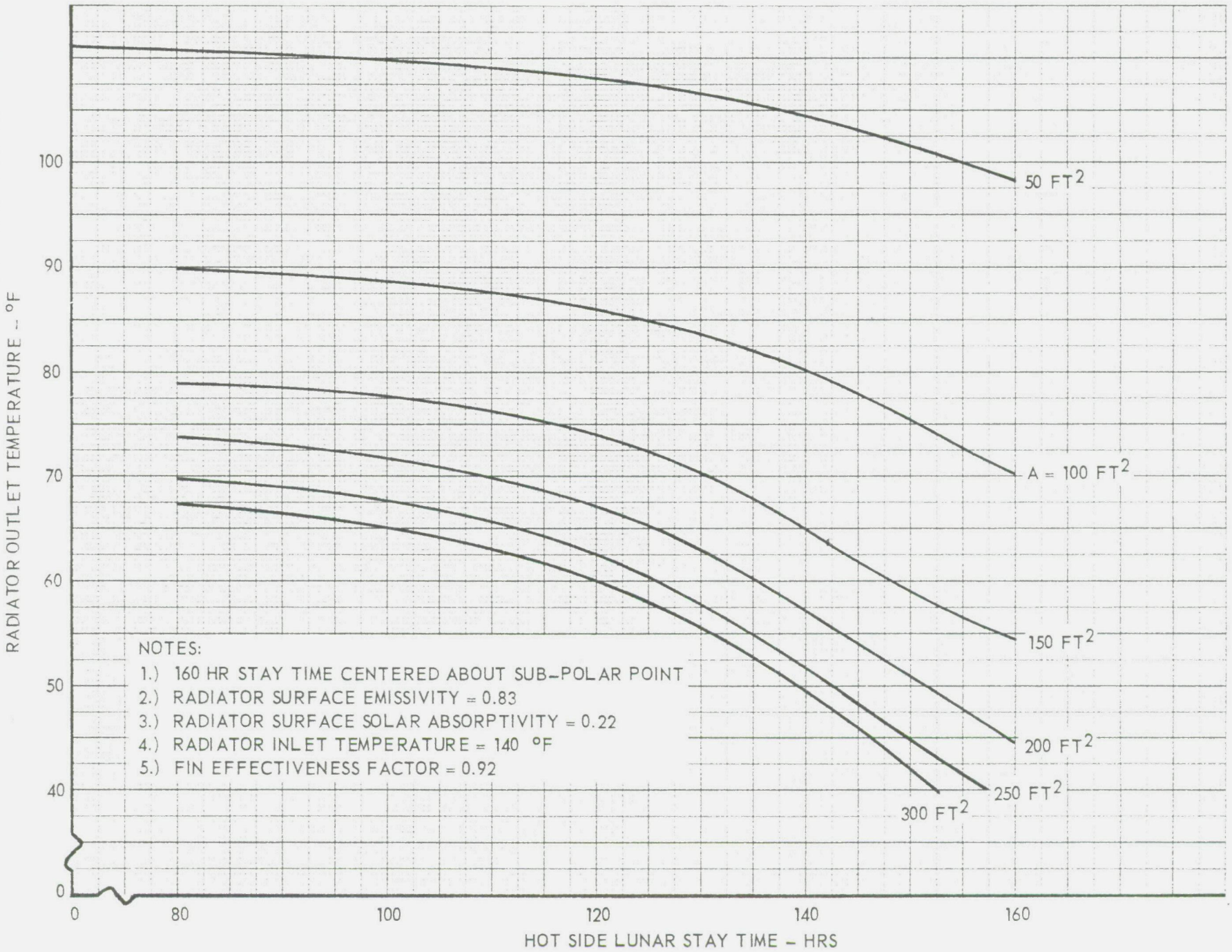


Figure 16–36 RADIATOR OUTLET TEMPERATURE VS. LUNAR HOT SIDE STAY TIME

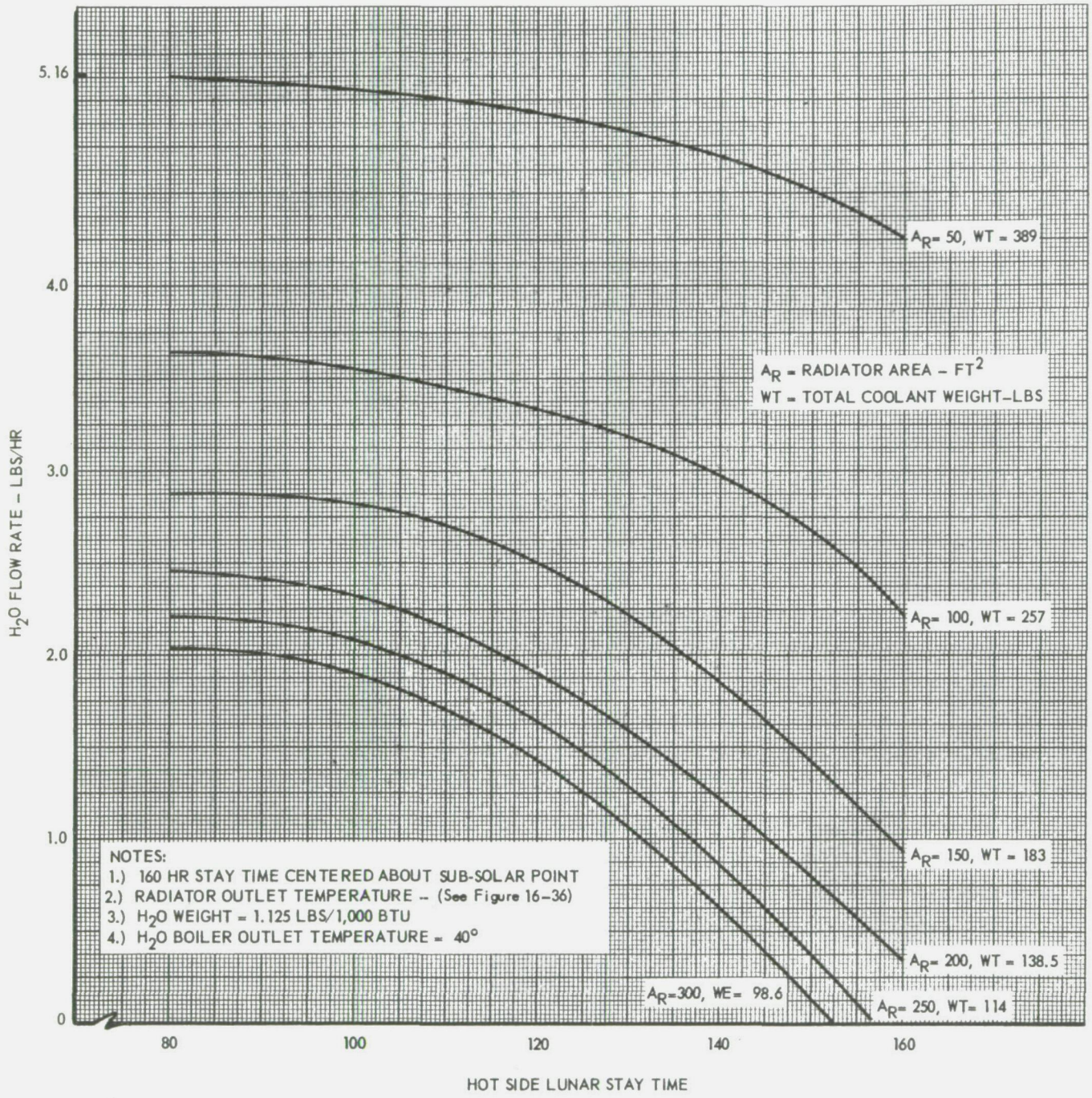


Figure 16-37 WATER BOILER FLOW RATE VS. LUNAR HOT SIDE STAY TIME

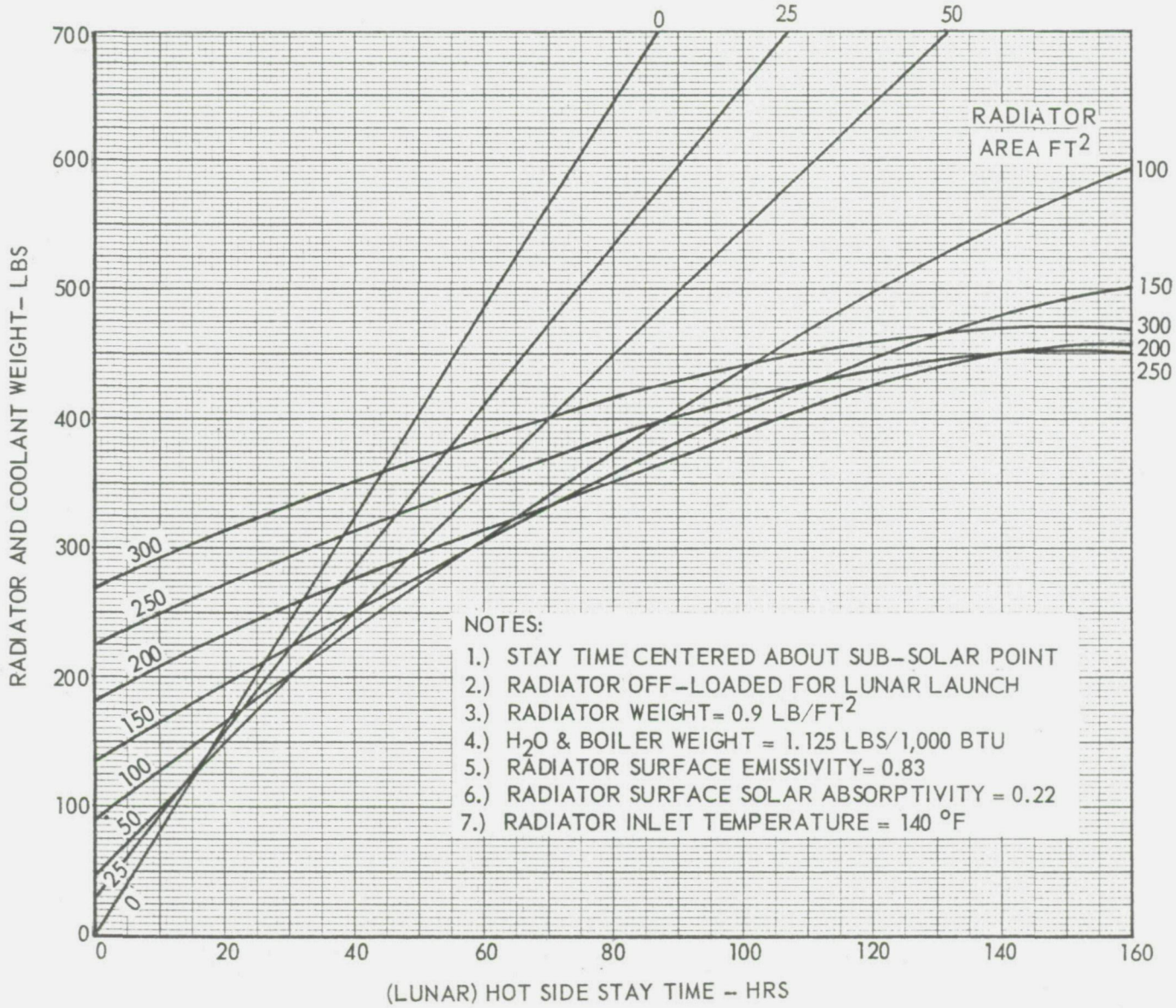


Figure 16-38 OPTIMUM WEIGHT (SPACE RADIATOR – H_2O BOILER COMBINATION) VS. HOT SIDE STAY TIME

NOTES:

1.) STAY TIME CENTERED ABOUT SUB-SOLAR POINT
2.) RADIATOR WEIGHT = 0.9 LBS/ FT^2
3.) H_2O BOILER WEIGHT = 1.125 LBS/1,000 BTU
4.) SOLAR ABSORPTIVITY = 0.22
5.) RADIATOR SURFACE EMISSIVITY = 0.83
6.) FIN EFFECTIVENESS = 0.92
7.) RADIATOR INLET TEMPERATURE = 140°F

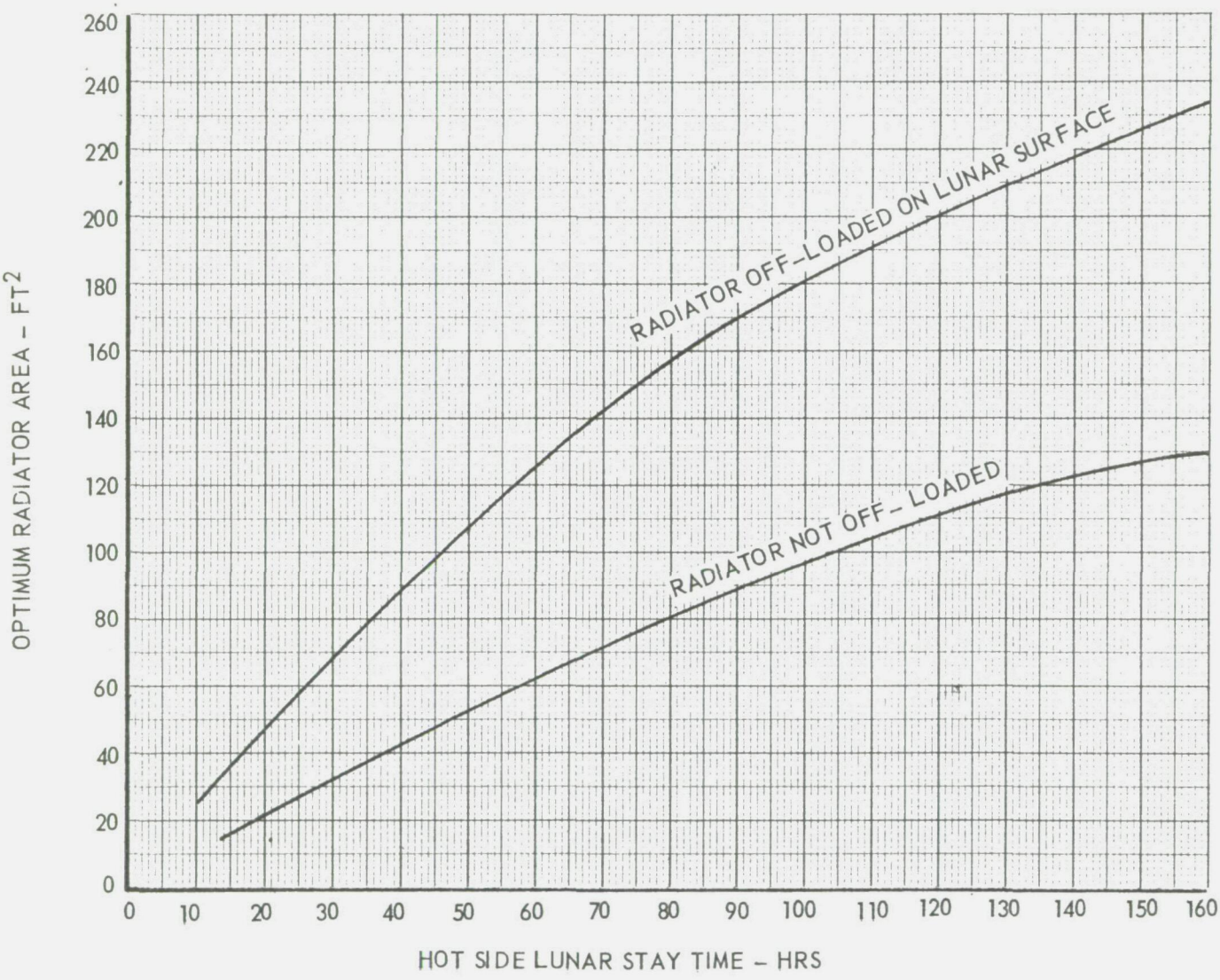


Figure 16-39 OPTIMUM RADIATOR AREA VS. LUNAR HOT SIDE STAY TIME

The study of an atmosphere control system using an analog requires that the functions of the system components be expressed as mathematical equations. From these equations, analogs are constructed which simulate the operating characteristics of each component. Use of the analog allows changing of components, control procedures, and operating variables with minimum effort. For example, a simple change to the analog setup will allow determination of the effect on the system of different cabin pressures, leakage rates, controls, control settings, airflows, temperatures, crew activity levels, cabin volumes, etc. With the analog, these data are obtained for the entire system as a closed loop. This type analysis provides the basis for selection of the most practical components, control procedures and system operating levels.

The equations, assumptions and results of the analog study are presented in Appendix A. Data from the analog runs were used to determine the performance of the system with different control procedures, airflows, crew activity levels, cabin pressures, cabin wall temperatures and leakage rates. These data were then used as the basis for selection of the recommended control procedures, system configuration and operating levels. The conclusions from the analog runs and the effect of these conclusions on the LEM environmental control system are as follows:

(a) For a constant air flow system, CO_2 concentration can be maintained within allowable limits without controls. The recommended CO_2 removal system utilizes an LiOH cannister with a constant cannister airflow. A cannister by-pass system is not provided.

(b) For a constant airflow system, relative humidity can be controlled within allowable limits by maintaining a constant coolant fluid inlet temperature to the suit circuit heat exchanger. This provides a relatively constant air temperature for water condensation and subsequent removal. This method is recommended for the LEM.

(c) A significant variation in the rate at which heat must be rejected from the cabin heat exchanger occurs as crew activity changes. The LEM cabin heat exchanger must be designed for this wide range.

(d) The cabin heat load established the minimum air flow through the atmosphere control system. A separate heat exchanger for cabin temperature control is desirable. The recommended LEM system incorporates a suit circuit heat exchanger for thermal control of the suit. A separate high airflow low pressure drop circuit is provided for cabin thermal control.

(e) The heat load is sensitive to changes in cabin wall temperature. Therefore, adequate insulation of the cabin wall is required. The recommended LEM vehicle incorporates 0.1 inches of "super insulation" on the outside surface.

RECOMMENDED ENVIRONMENTAL CONTROL SYSTEM

16.3.5 Recommended Environmental Control System Description

16.3.5.1 System Definition

The ECS provides thermal and atmosphere control of the LEM during lunar orbit, lunar landing, lunar rest, lunar launch and rendezvous with the command module. During the translunar flight, the ECS is inactive and the LEM is either unoccupied or is conditioned by the command module ECS.

The ECS of the LEM is designed for a crew of two men, a lunar cold side stay time of 24 hours and a flight time (descent and ascent) of 5.67 hours. The system provides for growth to an ultimate mission capability of seven days on the cold side of the lunar surface and two days on the hot side of the lunar surface. Extension of the mission capability does not involve major modifications as will be shown later.

The ECS consists of five major "loops" or circuits. These are: (1) the suit circuit, (2) the cabin circuit, (3) the thermal control loop, (4) the water loop, and (5) the oxygen supply loop. A schematic of the ECS showing the five loops is presented in Figure 16-40. Identifying numbers near each component refer to entries in Table 16-XII, which presents a detailed weight and volume breakdown of the ECS for the 24-hr lunar cold side mission. (Note-Figure 16-40 is reproduced as a fold-out on the last page of this volume in order to facilitate referral to the schematic while studying system operation)

The suit circuit controls the atmosphere composition and temperature of the pressure suits when the cabin is unpressurized. When the cabin is pressurized, the suit circuit remains in operation to remove water vapor, CO_2 and trace contaminates from the cabin atmosphere. The cabin circuit controls the cabin atmosphere temperature and provides circulation in the cabin. Heat removed from the suit and cabin circuits and the heat dissipated by all electronic equipment is transferred to the thermal control loop. The water loop provides storage for the water transported from earth and the water produced by the fuel cells. This water is used for crew consumption, crew sanitation, and cooling. The oxygen supply loop provides storage and delivery of the oxygen to control the pressure of the suit circuit and the cabin.

A portable life support unit (PLSU) is included for each crew member. The PLSU's provide environmental control for operation with pressure suits outside the spacecraft. In addition, the PLSU's provide emergency environmental control for the pressure suits in the event of a failure of the primary vehicle system. The PLSU's include expendables for four hours of operation which is sufficient time for a successful abort.

16.3.5.2 ECS Operation

16.3.5.2.1 General Operation

The ECS is thoroughly checked out prior to earth launch and then deactivated. During earth launch, the cabin pressure relief valve functions automatically and vents the LEM cabin pressure to 6 psia. After

Page intentionally left blank

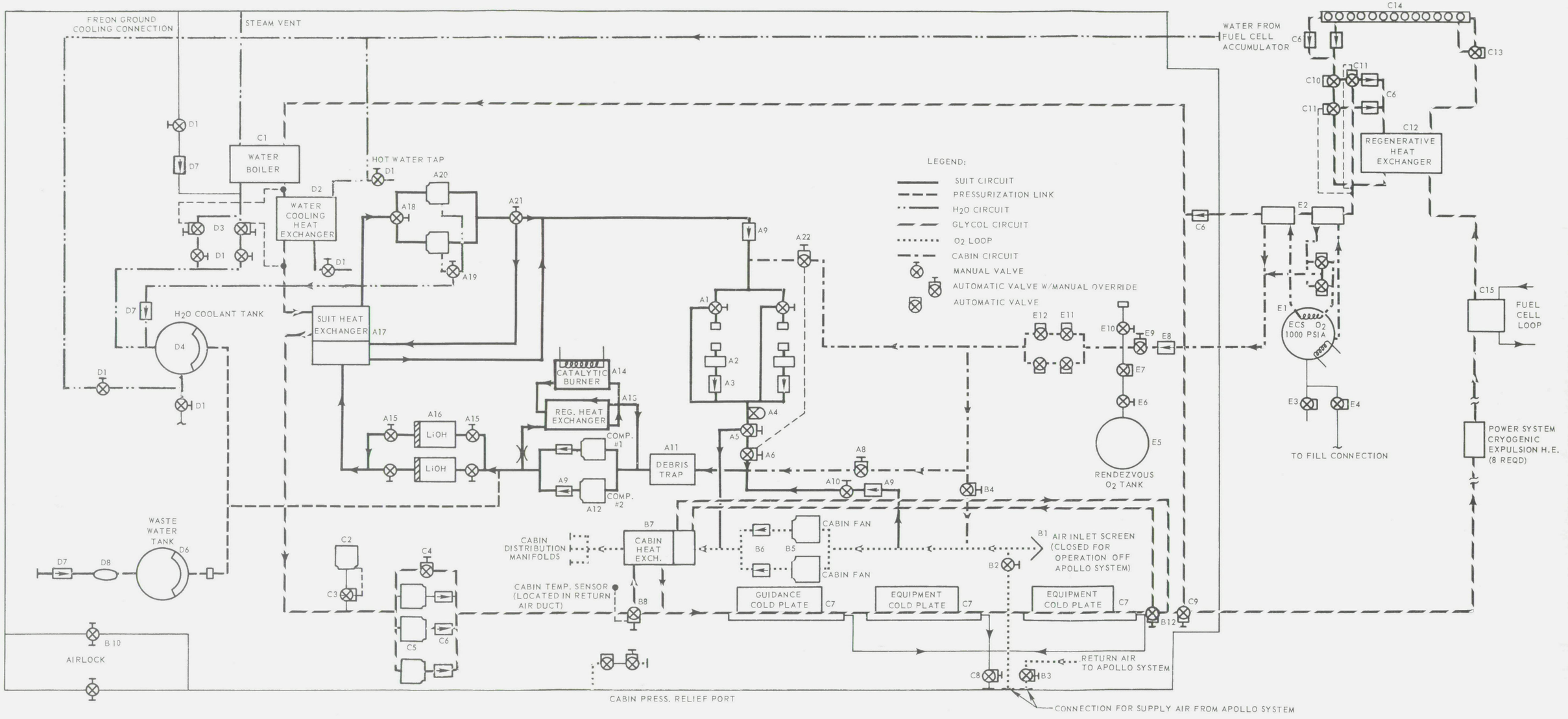


Figure 16–40 RECOMMENDED ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC

Page Intentionally Left Blank

TABLE 16-XII ENVIRONMENTAL CONTROL SYSTEM COMPONENT LIST

SUIT CIRCUIT COMPONENTS

ITEM NO.*	DESCRIPTION	24 HOUR COLDSIDE MISSION						
		QTY	VOL EA IN^3	VOL TOTAL IN^3	SIZE EA IN.	WEIGHT – LBS DRY EA	WEIGHT – LBS FLUID EA	WEIGHT – LBS TOTAL
A-1	SUIT DIVERTER VALVE	2	50	100	4 x 4 x 3	2.4	–	4.8
A-2	DEBRIS & VOMIT TRAP ,P	2	29	58	3D x 4	1.2	–	2.4
A-3	SUIT CHECK VALVE	2	16	32	3.2D x 2	0.3	–	0.6
A-4	SUIT CIRCUIT RELIEF VALVE	1	3	3	1 x 3	1.1	–	1.1
A-5	DIVERTER VALVE, BAROMETRIC	1	36	36	3D x 4	1.1		1.1
A-6	EMERGENCY SHUT-OFF VALVE	1	36	36	3D x 4	1.6		1.6
A-7	NOT USED							
A-8	CABIN PRESSURE CONTROL VLV	1	36	36	3 x 3 x 4	1.6		1.6
A-9	SUIT CIRCUIT CHECK VLV	4	16	64	3.2Dx2	0.3		1.2
A-10	AIR INLET SHUT-OFF VLV	1	12	12	3.2Dx3	0.6		0.6
A-11	DEBRIS TRAP	1	157	157	5D x 8	2.5		2.5
A-12	SUIT CIRCUIT COMPRESSOR	2	360	720	7.5Dx8	8.0		16.0
A-13	REG. HEAT EXCHANGER	1	PART OF A-14 BELOW					
A-14	CATALYTIC BURNER	1	98	98	5D x 5	2.0	–	2.0
A-15	LIOH BED SELECTOR VL V	4	12	48	3.2Dx3	0.6		2.4
A-16	CO_2 & ODOR REMOVAL CANISTER	2	704	704	10 D x 9	10.4	6.5	16.9
A-17	SUIT CIRCUIT HEAT EXCH.	1	504	504	8 x 7 x 9	6.0	–	6.0
A-18	H_2O SEPARATOR INLET SELECTOR VL V	1	12	12	3.2Dx3	0.9	–	0.9
A-19	H_2O SEPARATOR OUTLET SELECTOR VL V	1	8	8	2 x 2 x 2	0.4		0.4
A-20	H_2O SEPARATOR	2	156	312	5.5D x 6.5	3.4		6.8
A-21	SUIT TEMP. CONTROL BYPASS VLV	1	12	12	3.2Dx4	0.9		0.9
A-22	EMERGENCY O_2 FLOW VALVE	1	3	3	1D x 3	1.1		1.1
	TOTALS			2948				70.6

CABIN CIRCUIT COMPONENTS

ITEM NO.*	DESCRIPTION	24 HOUR COLDSIDE MISSION						
		QTY	VOL EA IN^3	VOL TOTAL IN^3	SIZE EA IN.	WEIGHT – LBS DRY EA	WEIGHT – LBS FLUID EA	WEIGHT – LBS TOTAL
B-1	CABIN AIR INLET SCREEN	1	8	8	4 x 4 x ½	0.8	–	0.8
B-2	EXTERNAL SUPPLY INLET VALVE	1	59	59	5Dx3	1.1	–	1.1
B-3	EXTERNAL SUPPLY RETURN VLV	1	59	59	5Dx3	1.1	–	1.1
B-4	CABIN RE-PRESSURIZATION VALVE	1	3	3	1Dx3	0.4		0.4
B-5	CABIN FANS	2	550	1100	10Dx7	5.6		11.2
B-6	CABIN FAN CHECK VLV	2	39	78	5Dx2	0.5		1.0
B-7	CABIN HEAT EXCHANGER	1	243	243	3 x 9 x 9	5.1		5.1
B-8	CABIN TEMPERATURE CONTL VLV	1	12	12	2 x 2 x 3	2.0		2.0
B-9	PRESSURE RELIEF SHUT-OFF VLV	1	59	59	5D x 3	1.1		1.1
B-10	AIRLOCK VALVES	2	50	100	4D x 4	2.1		4.2
B-11	PRESSURE RELIEF VLV	1	485	485	9 x 9 x 6	5.0		5.0
B-12	CABIN HEAT CONTROL VALVE	1	6	6	3 x 2 x 1	0.5		0.5
	TOTALS			2212				33.5

THERMAL LOOP COMPONENTS

ITEM NO.*	DESCRIPTION	24 HOUR COLDSIDE MISSION						
		QTY	VOL EA IN^3	VOL TOTAL IN^3	SIZE EA IN.	WEIGHT – LBS. DRY EA	WEIGHT – LBS. FLUID EA	WEIGHT – LBS. TOTAL
C-1	WATER BOILER	1	128	128	8 x 4 x 4	4.0		4.0
C-2	GLYCOL ACCUMULATOR	1	400	400	5D x 11	3.2	8.0	11.2
C-3	ACCUMULATOR SHUT-OFF VALVE	1	2	2	1D x 2	0.3	–	0.3
C-4	GLYCOL PUMP BY-PASS VALVE	1	6	6	3 x 2 x 1	0.5	–	0.5
C-5	GLYCOL PUMP	3	38	114	3.5D x 4	1.8	–	5.4
C-6	GLYCOL CHECK VALVE	7	1	7	1D x 1	0.2	–	1.4
C-7	COLD PLATES	1 - 3	1080	1080	10' x 10' x ¾	30.3	34.3	64.6
C-8	EMERGENCY COOLING VALVE	1	12	12	2 x 2 x 3	0.4	–	0.4
C-9	RADIATOR BY-PASS VALVE	1	6	6	3 x 2 x 1	0.5	–	0.5
C-10	CIRCUIT SELECTOR VALVE	1	6	6	3 x 2 x 1	0.5	–	0.5
C-11	GLYCOL MODULATING VALVE	2	18	36	2.5D x 3	1.1	–	2.2
C-12	REGENERATIVE HEAT EXCHANGER	1	125	125	5 x 5 x 5	4.2		4.2
C-13	RADIATOR CIRCUIT SELECTOR VLV	1	6	6	3 x 2 x 1	0.5		0.5
C-14	RADIATOR	1	–	–	(35 ft2)	31.5		31.5
C-15	FUEL CELL HEAT EXCHANGER	1	144	144	16 x 3 x 3	5.0		5.0
	TOTALS			2072			42.3	132.2

*ITEM NUMBERS REFER TO NUMBERS ON ECS SCHEMATIC (Figure 16-40)

Page Intentionally Left Blank

TABLE 16-XII ENVIRONMENTAL CONTROL SYSTEM COMPONENT LIST (CONT'D.)

WATER MANAGEMENT LOOP COMPONENTS

ITEM NO. *	DESCRIPTION	24 HOUR COLDSIDE MISSION						
		QTY	VOL EA IN^3	VOL TOTAL IN^3	SIZE EA IN.	WEIGHT – LBS DRY EA.	WEIGHT – LBS FLUID EA.	WEIGHT – LBS TOTAL
D-1	WATER SHUT-OFF VALVES	6	8	48	2 x 2 x 2	0.3	–	1.8
D-2	WATER COOLING HEAT EXCHANGER	1	64	64	4 x 4 x 4	2.0		2.0
D-3	WATER FLOW CONTROL VALVE	2	8	16	2 x 2 x 2	1.1		2.2
D-4	COOLING WATER TANK	1	1450	1450	14D Sphere	8.8	53.3	62.1
D-5	(NOT USED)							
D-6	WASTE WATER TANK	1	500	500	10D Sphere	5.1		5.1
D-7	WATER CHECK VALVES	3	3	9	2D x 1	0.1		0.3
D-8	URINE ION-EXCHANGE FILTER	1	105	105	4D x 7	6.0		6.0
	TOTALS			2192			53.3	79.5

*ITEM NUMBERS REFER TO NUMBERS ON ECS SCHEMATIC (FIGURE 16-40)

OXYGEN STORAGE SYSTEM COMPONENTS

ITEM NO *	DESCRIPTION	24 HOUR COLDSIDE MISSION						
		QTY	VOL EA IN^3	VOL TOTAL IN^3	SIZE EA IN.	WEIGHT – LBS DRY EA	WEIGHT – LBS FLUID EA	WEIGHT – LBS TOTAL
E-1	NORMAL O_2 STORAGE TANK	1	1520	1520	14.3D SPH.	30	50.0	80.0
E-2	CRYOGENIC EXPULSION HEAT EXCHANGER	2	2	4	1D x 2	1.0	–	2.0
E-3	O_2 VENT VALVE	1	12	12	2 x 2 x 3	0.4		0.4
E-4	O_2 FILL VALVE	1	12	12	2 x 2 x 3	0.3		0.3
E-5	RENDEZVOUS O_2 TANK	1	220	220	7.6D SPH	8.9	4.0	12.9
E-6	O_2 TANK SHUT OFF VALVE	1	12	12	2 x 2 x 3	0.4		0.4
E-7	O_2 PRESS REDUCER	1	12	12	2 x 2 x 3	0.9		0.9
E-8	O_2 CHECK VALVE	1	1	1	1D x 1	0.2		0.2
E-9	NORMAL O_2 SHUT OFF VALVE	1	12	12	2 x 2 x 3	0.4		0.4
E-10	BACKPACK FILL VALVE	1	12	12	2 x 2 x 3	0.3		0.3
E-11	O_2 PRESSURE REDUCER	2	12	24	2 x 2 x 3	0.8		1.6
E-12	O_2 PRESSURE REDUCER	2	12	24	2 x 2 x 3	0.8		1.6
	TOTAL			1865			54.0	101.0

WEIGHT AND VOLUME SUMMARY

ITEM NO	DESCRIPTION	WT. TOTAL LB.	VOL TOTAL IN 3
A	SUIT CIRCUIT	70.6	2948
B	CABIN CIRCUIT	33.5	2212
C	TERMAL LOOP	132.2	2072
D	WATER LOOP	79.5	2192
E	O_2 SUPPLY LOOP	101.0	1865
	TOTAL	433.0	11289
	ELECTRICAL HARDWARE	21.7	564
	MECHANICAL HARDWARE	65.0	1693
	TOTAL	503.5	13546

Page Intentionally Left Blank

injection into a lunar trajectory and before the first mid-course correction, the LEM is transferred to the front of the command module. (See Section 3 Volume II - "Docking"). The large window area and large free volume available in the LEM provides a desirable place for the crew to rest and make observations. In order to provide a conditioned atmosphere in the LEM during the translunar phase at a minimum overall penalty, the command module ECS is utilized for LEM thermal and atmosphere control. This is accomplished by supply and return ducts connecting the LEM air distribution system to the command module ECS. After docking is completed, the command module ECS conditions both vehicles for the remainder of the translunar phase.

When the spacecraft is injected into a lunar orbit, two crew members enter the LEM and begin pre-landing checkout of the electronics systems. At this time the LEM thermal control loop is activated. The LEM radiator is not filled prior to this time in order to prevent freezing during the translunar flight. An alternate method to prevent radiator freezing during the translunar phase is to connect the LEM thermal control loop with the command module thermal control loop. Sufficient glycol from the command module system could be circulated through the LEM radiator to prevent freezing. The main disadvantage of this alternate method is the undesirability of the additional fluid connections between the two spacecraft.

During pre-landing checkout, the LEM supercritical oxygen storage tank is brought to operating pressure by an electrical heater within the tank. After the initial pressurization, an integral pressurization system with a glycol heat exchanger maintains the tank at the proper pressure. (Figure 16-40).

Immediately prior to separation for lunar descent, the pressure suits are connected to the suit circuit and the LEM cabin is depressurized. As the cabin pressure decreases below 5.0 psia, the suit circuit is automatically purged of the oxygen-nitrogen atmosphere supplied by the command module. The cabin remains depressurized during lunar descent and landing to prevent the possibility of an explosive decompression should the LEM be damaged. Once the LEM has landed on the lunar surface, the cabin is pressurized and remains pressurized during lunar rest. Egress and entry to the LEM are provided through an air lock.

In order to decrease the weight of the LEM for lunar launch, the cryogenic oxygen tankage, expended LiOH cannisters, etc., are off-loaded on the lunar surface. A separate gaseous oxygen tank supplies oxygen for the rendezvous phase of the mission.

In the event of a complete failure of the atmosphere control system, the portable life support units provide atmosphere control for the crew during abort. Should a failure of the thermal control system occur, heat can be dissipated from electrical and electronic equipment by water wicking coolers which are built integrally with the cold plates. This emergency cooling system is activated by opening a valve allowing the water to boil and vent to space.

16.3.5.2.2 Atmosphere Control Operation

Due to the many mission phases and reliability requirements, the atmosphere control system must provide various basic modes of operation. The following paragraphs present a brief description of both the basic modes of operation and the more significant capabilities of the atmosphere control system.

Leak Check of the Suit Circuit

Prior to depressurization of the cabin for lunar descent or lunar launch, a leak check must be performed on the suit circuit. A portion of the ECS schematic (Figure 16-40) is reproduced in Figure 16-41, to show the leak check operation. The diverter valve (A-5 in Figure 16-41) is automatically opened to the cabin circuit by a cabin pressure greater than 4.6 psia. For the leak check, the suit circuit diverter valve (A-5) is manually positioned to isolate the suit circuit from the cabin circuit. The emergency shut-off valve (A-6) is normally open and oxygen is free to circulate within the suit circuit. The emergency oxygen supply valve (A-22) is then manually opened and allowed to remain open until the suit pressure reaches approximately 5 psi above the cabin pressure. The valve is then shut-off and the leakage is determined by the rate of decay of the suit circuit pressure. After the leakage rate has been determined to be satisfactory, the diverter valve (A-5) is manually opened allowing the suit circuit pressure to equalize with the cabin pressure.

Purge of Nitrogen with the LEM

Prior to decompression for lunar descent the LEM cabin is filled with a 7.0 psia atmosphere of oxygen and nitrogen from the command module. Since the LEM system provides a 5.0 psia pure oxygen atmosphere, the nitrogen must be purged from the cabin and suit circuits. The nitrogen purge is accomplished automatically during decompression of the cabin. Figure 16-42 shows the system operation during the purge. After the pressure suits are connected to the suit circuit, the cabin dump switch is placed in the "dump" position. This action opens the cabin pressure relief valve, which vents the cabin gas to space. When the suit circuit pressure drops to 5.0 psia, the oxygen supply valve (A-8) opens automatically and allows pure oxygen to flow into the suit circuit. The oxygen passes through the entire suit circuit and enters the cabin circuit downstream of the diverter valve (A-5). The cabin decompression flow rate is larger than the oxygen make-up flow rate. Therefore, both the suit circuit and cabin pressures continue to decrease. When the cabin pressure reaches approximately 4.6 psia, the diverter valve (A-5) automatically closes the vent to the cabin circuit and allows the oxygen to circulate within the suit circuit. Sufficient pure oxygen to completely purge the suit circuit is introduced prior to the time when the cabin pressure drops to 4.6 psia. The oxygen make-up flow continues until the suit circuit pressure builds back up to 5.0 psia.

Suit Circuit Atmosphere Control With An Unpressurized Cabin

As previously noted, the cabin is unpressurized during lunar descent, lunar launch and rendezvous and atmosphere control of the pressure

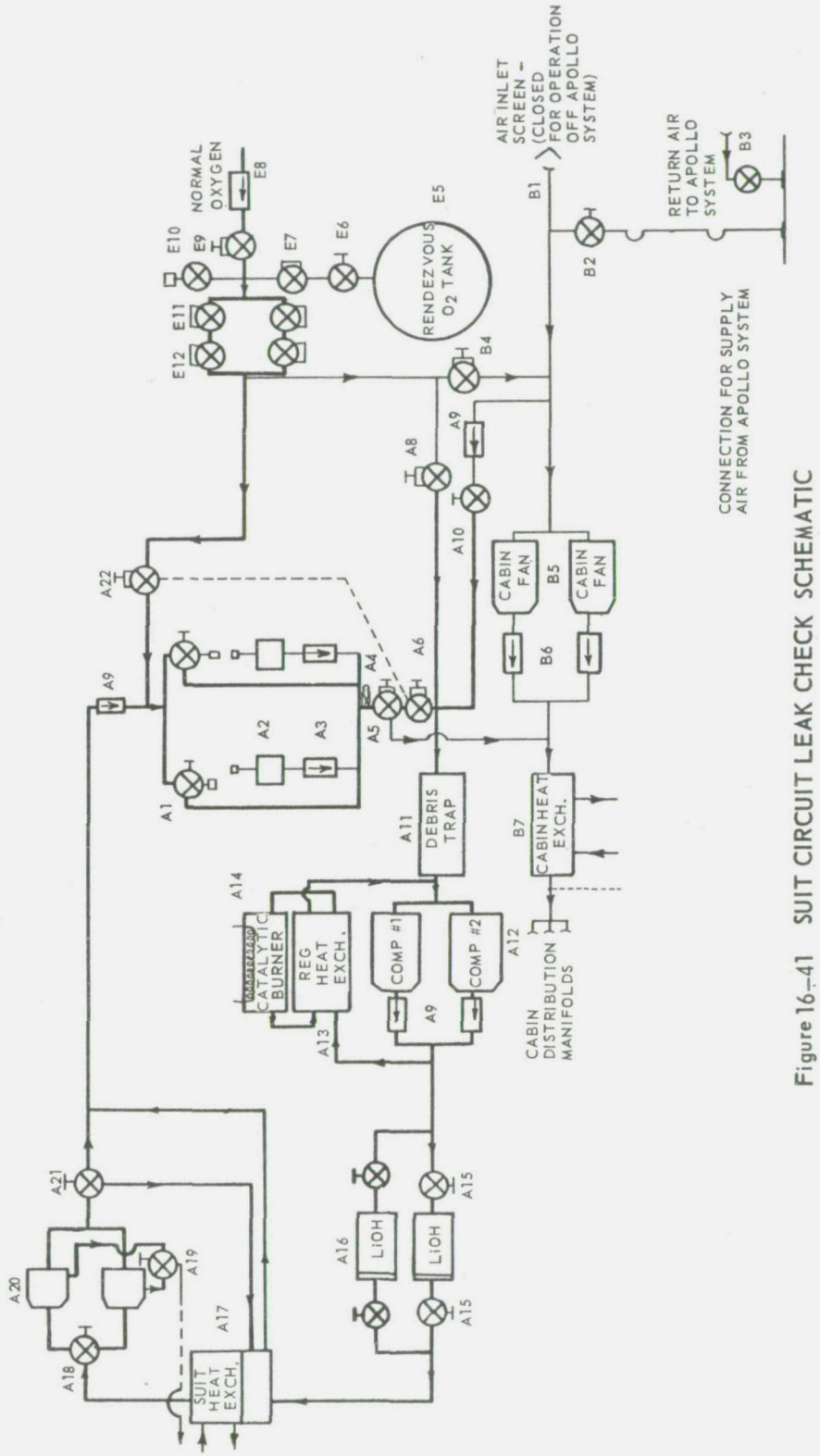


Figure 16-41 SUIT CIRCUIT LEAK CHECK SCHEMATIC

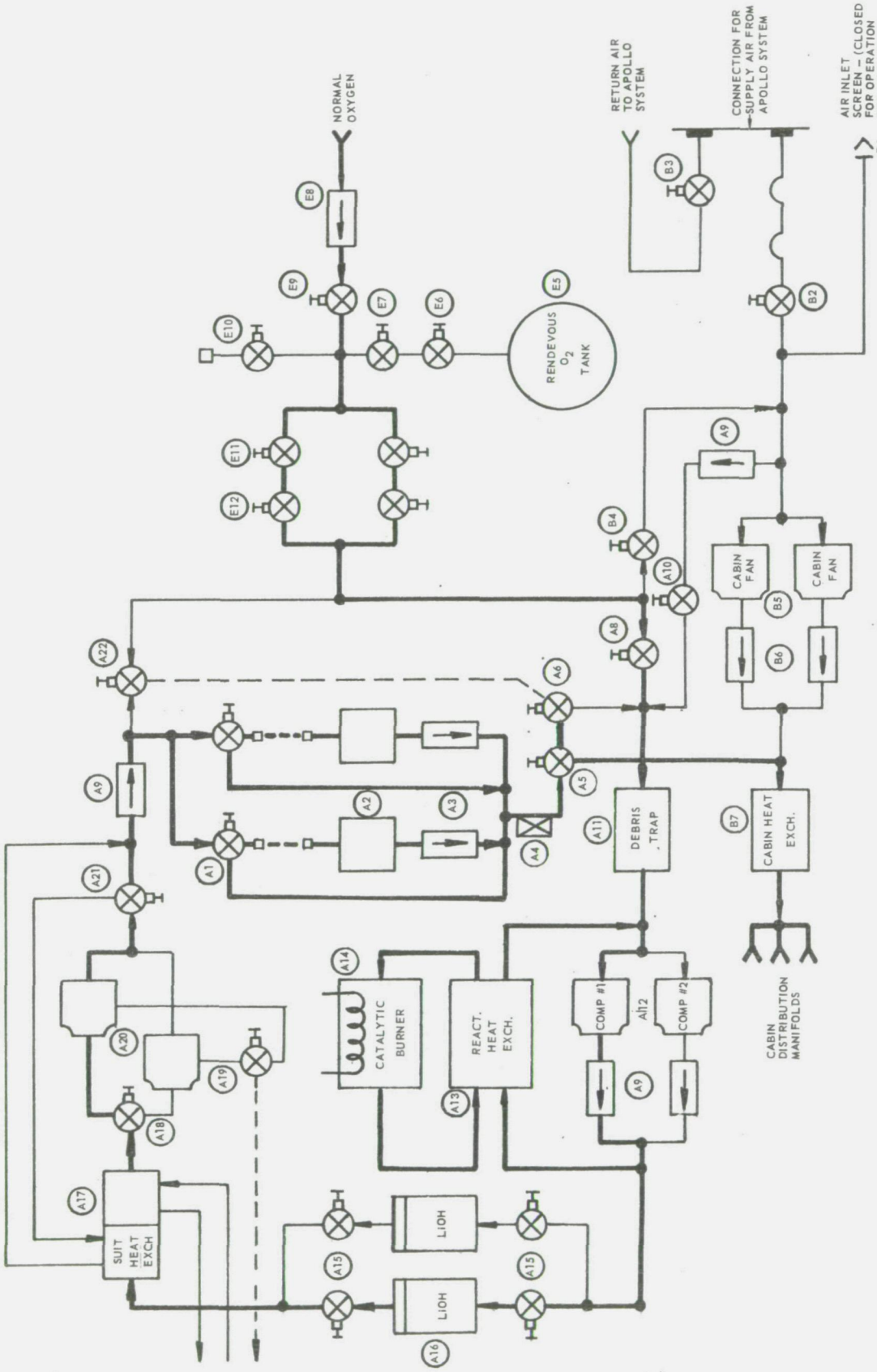


Figure 16-42 SUIT CIRCUIT PURGE SCHEMATIC

suits is provided by the suit circuit. Operation of the suit circuit with an unpressurized cabin is shown by Figure 16-43. During unpressurized cabin operation, oxygen is supplied from the normal ECS oxygen tank to the pressure reducing valves (E-11 and E-12). These four valves are in a series-parallel arrangement,and failure of any one valve (open or closed) will not affect system operation. Oxygen make-up flow through the pressure regulator (A-8) controls the pressure in the suit circuit to 5.0 psia. This valve has two parallel passages with individual barometric pressure sensors for redundancy. A manual selector control provides "off", "passage one", and "passage two" selection.

As a safety feature during the lunar landing maneuver, the rendezvous oxygen system is designed to supply immediate oxygen flow in the event of a failure of the normal oxygen supply. The rendezvous oxygen tank shut-off valve (E-6) is opened prior to initiation of lunar descent. The pressure reducer (E-7) is designed to regulate the rendezvous oxygen pressure to 800 psia. As long as the normal oxygen supply maintains the design pressure of 1,000 psia, no oxygen will flow from the rendezvous oxygen tank. Should the normal oxygen system fail and allow the pressure to drop below 800 psia, the rendezvous tank will maintain oxygen flow to the suit circuit. The pressure reducing valve (E-7) has two parallel passages so that flow will not be disrupted should one passage fail closed. The valves E-8, E-9, E-10, E-11 and E-12 are designed to withstand pressures up to 7500 psia (the rendezvous gas storage pressure) so that failure of either passage of valve E-7 open will not overpressurize the oxygen system.

Since the normal oxygen supply tank is off-loaded on the lunar surface, only the rendezvous oxygen supply is available for lunar launch and rendezvous. Operation of the atmosphere control system is identical to the descent mode of operation except for the oxygen supply. The normal oxygen supply shut off valve (E-9) is closed and oxygen is drawn from the rendezvous tank as required to maintain a 5 psia pressure in the suit circuit.

The oxygen flow is circulated through the suit circuit at a rate of 30 cfm. The oxygen passes through a debris trap (A-11) prior to entry into the suit compressors. The debris trap removes solid matter picked up by the oxygen flow. Should the debris trap become clogged, a relief-bypass arrangement opens so that the flow is not terminated.

Two compressors (A-5) are provided for circulation of oxygen flow through the suit circuit. Each compressor is capable of providing the required flow and only one compressor is operated at any one time. A switch on the control panel is used to turn either compressor on or to turn both off. Either compressor will develop a head of approximately 17 inches water to overcome the pressure drop in the suit circuit.

A catalytic burner (A-14) is provided to remove trace contaminants. Approximately 10% of the suit circuit flow passes through the regenerative heat exchanger (A-13) catalytic heater (A-14) and then back into the suit circuit. The flow through the catalytic burner is continuous although the burner is normally inactive. The catalytic burner is activated manually when a build-up of contaminants is indicated by the gas analyzer or by smell. The catalytic burner is not required for the 24-hour mission but

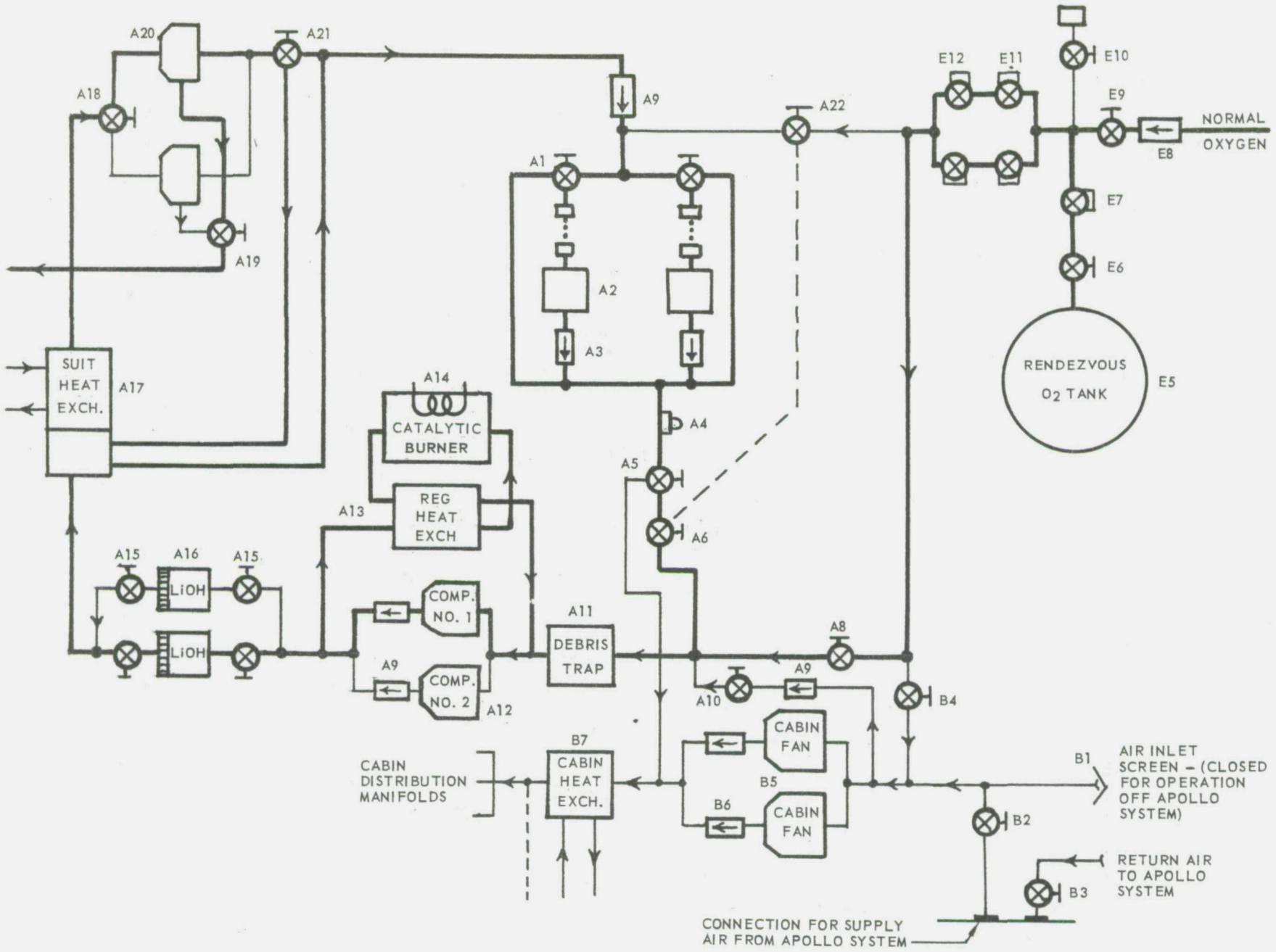


Figure 16–43 SUIT CIRCUIT OPERATION WITH AN UNPRESSURIZED CABIN

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

has been included in the system to provide growth for the extended mission capability.

One lithium hydroxide canister for CO_2 removal is provided for each 24 hours of mission duration. An activated charcoal filter is built into each canister for odor removal. Two canister housings (A-16) each designed to accept an expendable LiOH and activated charcoal charge are provided. Isolation valves are provided so that one canister is used while the other is on standby. The two housings are provided so that spent canisters can be replaced without disrupting flow through the suit circuit. For the 24-hour mission the LiOH and activated charcoal charge in each canister would be reduced.

Temperature and humidity control of the atmosphere in the suit circuit is accomplished by the suit circuit heat exchanger (A-17) and the water separator (A-20). The oxygen-water vapor mixture in the suit circuit is cooled in the suit heat exchanger by glycol from the thermal loop. The glycol enters the heat exchanger at 40^{o} F and cools the mixture to approximately 45^{o}F. This low temperature condenses moisture in the oxygen. The oxygen and water mixture then enters a centrifugal water separator where the liquid water is separated and transferred into the water tank. The separator is driven by the oxygen flow in the system. Two separators are provided for redundancy and only one is used at any one time. After leaving the water separator, part of the oxygen flow is diverted back through the suit heat exchanger (A-17) where it is reheated by the warm air flow. The amount of oxygen flow diverted determines the suit inlet temperature. The fraction of flow diverted may be adjusted manually by valve A-21 to control the suit inlet temperature between approximately 40^{o}F and 80^{o}F.

The oxygen flow through each suit is controllable independently between 5 and 15 cfm. The suit diverter valves (A-1) are used to bypass part of the flow around each suit. Built-in flow restrictions in the bypass provide a relatively constant pressure drop across the pressure suits regardless of the amount of flow bypassed. When the pressure suits are not connected, all of the flow is bypassed. Replaceable vomit traps (A-2) are installed downstream of the suits to prevent contamination of the entire circuit should one of the crew become sick.

Suit and Cabin Atmosphere Control With a Pressurized Cabin

Suit circuit operation with a pressurized cabin is similar to operation with an unpressurized cabin except that the oxygen flow is passed through the cabin rather than recirculated within a closed loop. The oxygen flow for this operation is shown in Figure 16-44. The diverter valve (A-5) is positioned so that the suit flow is vented into the cabin circuit and passes into the cabin through the cabin distribution ducts. Suit circuit oxygen flow is drawn from a common inlet with the cabin circuit. An inlet flow deflector and screen (B-1) prevents debris and liquids from entering the suit and cabin circuits.

When the cabin is pressurized, the cabin circuit provides temperature control and air circulation. Two cabin fans (B-5) provide flow through the cabin heat exchanger (B-7). Only one fan operates at any one time. Each fan has the capability to supply approximately 140 cfm (in addition to the suit

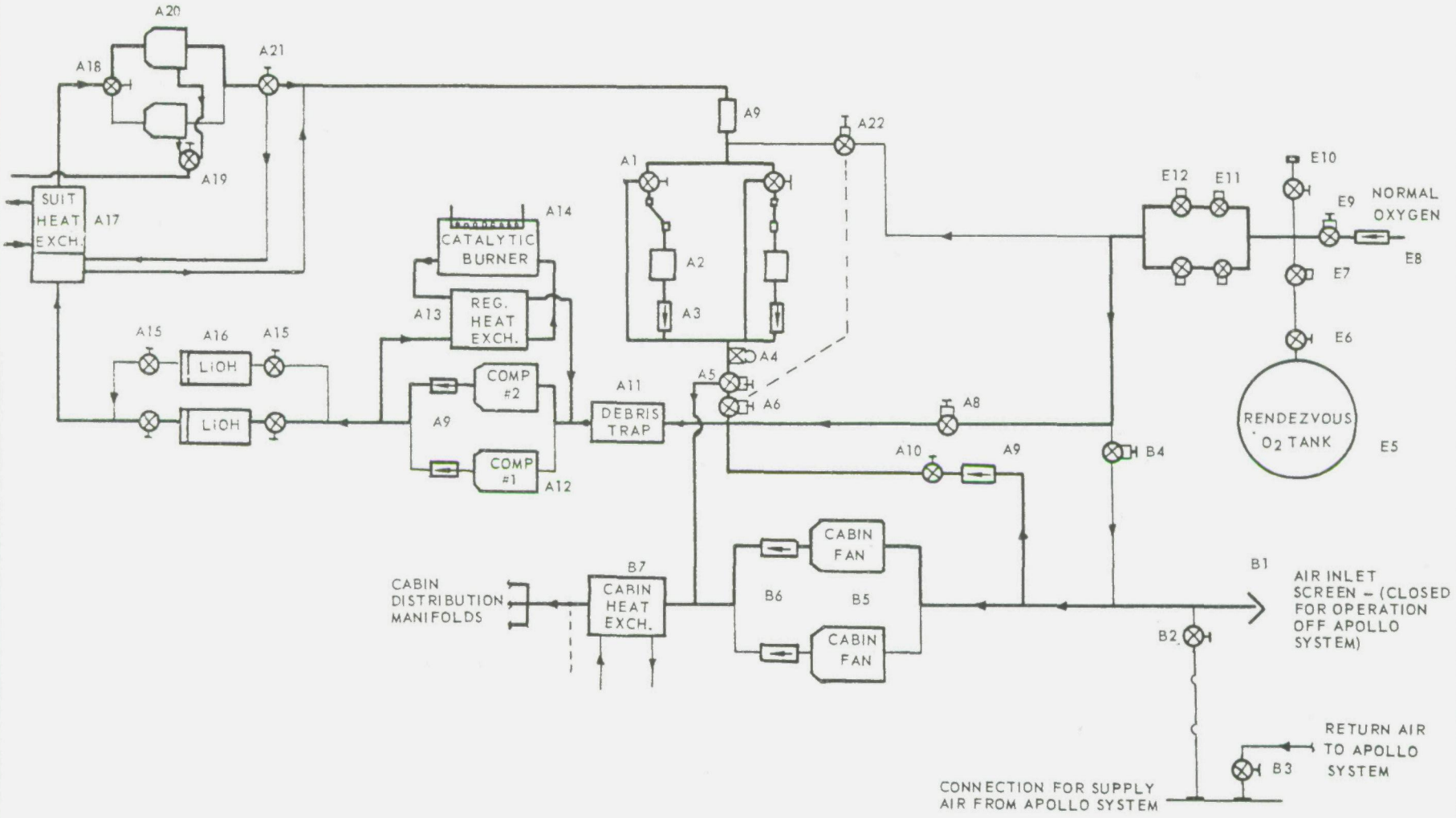


Figure 16–44 SUIT AND CABIN CIRCUIT OPERATION WITH PRESSURIZED CABIN

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

circuit flow) with a head of approximately 2 inches of water. The cabin temperature is maintained at 72 ± 4°F by diverting part of the glycol in the thermal loop through the cabin heat exchanger.

Emergency Oxygen Supply, Pressurized Cabin

In the event of a failure of the suit circuit, oxygen can be provided directly to the suits from the oxygen storage tanks for breathing, pressurization and cooling. The emergency oxygen supply valve (A-22) may be opened manually to provide oxygen flow for pressurization and cooling of the pressure suits. The diverter valve (A-5) vents the oxygen from the suit circuit into the cabin. As the cabin pressure builds up to 6.0 psia, the cabin pressure relief valve (B-11) opens allowing the excess gas to be vented to space. The check valve (A-9) prevents back flow through the suit circuit. The emergency oxygen supply valve (A-22) remains open until it is closed manually.

Emergency Oxygen Supply, Unpressurized Cabin

When the cabin is unpressurized, the diverter valve (A-5) is positioned to allow oxygen flow from the pressure suits through the suit circuit. Should the suit circuit pressure drop below 3.5 psia, the emergency oxygen supply valve (A-22) automatically opens. The emergency shut-off valve (A-6) receives a signal from the emergency oxygen supply valve and is closed by a solenoid. The pressure in the pressure suits then builds up until it reaches 6.0 psi above the cabin pressure. At this pressure the suit circuit pressure relief valve (A-4) opens. The oxygen flow then continues through the suits and out to the cabin through the relief valve (A-4).

Airlock Operation

An airlock is provided for entering and leaving the LEM without depressurizing the cabin. The airlock is pressurized by opening a pressurization valve (B-10 in Figure 16-40) to allow cabin air to flow into the airlock. Pressurization of the airlock requires approximately 5 minutes. Since the oxygen flow required to pressurize the airlock in this time period is greater than the make-up oxygen flow, the cabin pressure will drop slightly. The maximum cabin pressure drop during an airlock pressurization is approximately 0.4 psia. When the cabin and airlock pressures are equalized, the crew member enters the airlock and closes the valve to the cabin. The airlock is then vented by opening a depressurization valve (B-10) which allows the airlock to vent to space. The airlock pressurization and depressurization valves can be operated from either inside or outside the airlock.

Translunar Operation

During translunar flight, air is circulated through the LEM cabin circuit by the command module environmental control system. The ECS of the LEM is inactive during this time. Supply and return ducts are connected between the LEM and the command module and the shut-off valves (B-2 and B-3) are opened. The crew will normally not be wearing pressure suits in the LEM during the translunar flight. When pressure suits are worn, environmental control will be provided by the portable life support units.

Normal Oxygen Storage and Supply

The normal oxygen supply is stored in the supercritical oxygen storage tank (E-1). The tank is filled during pre-earth-launch servicing with liquid oxygen at atmospheric pressure. Immediately prior to launch the fill valve is closed. As heat is absorbed by the oxygen during the translunar flight, the tank pressure increases. Sufficient tank insulation is provided to preclude excessive oxygen pressures prior to the time the ECS is activated in lunar orbit. As noted previously the oxygen is pressurized to the supercritical operating pressure of 1,000 psia during checkout in lunar orbit. This is accomplished by a 500 watt electrical heater in the oxygen tank.

Once the tank has been pressurized, an automatic pressurization system maintains the pressure at 1,000 psia. As the fluid is withdrawn from the oxygen tank, it passes through a heat exchanger (E-2) and is heated to approximately 40°F. by the glycol loop. One of two regulating valves bypasses part of the flow back through a heat exchanger in the oxygen tank. Sufficient flow is bypassed to maintain the tank operating pressure of 1,000 psia. After passing through the storage tank and prior to entry into the cabin, the flow is heated to 40°F. in a second heat exchanger (E-2).

16.3.5.2.3 Thermal Control Loop Operation

A schematic of the thermal control loop is included in Figure 16-40. A 62.5% aqueous ethylene glycol heat transport fluid transports excess heat to the radiator (C-14) and/or water boiler (C-1) for dissipation to space. During operation on the cold side of the lunar surface the excess heat is dissipated entirely by the space radiator. Heat is transferred from the suit circuit and cabin circuit to the glycol in the suit circuit heat exchanger (A-17) and the cabin heat exchanger (B-7), respectively. The glycol flow to the suit circuit heat exchanger is controlled to 40°F by the water boiler and space radiator controls. Control of the glycol to this temperature provides the required suit circuit atmosphere exit temperature of 45°F. The amount of cooling provided by the cabin heat exchanger (when the cabin is pressurized) is controlled by varying the glycol flow through the heat exchanger (B-7). The glycol bypass valve (B-8) senses cabin temperature and maintains the required cold glycol flow rate through the heat exchanger. In the event the cabin temperature drops below 68°F, the cabin temperature warning light goes "on". The diverter valve (B-12) is then operated manually to divert part of the hot glycol flow (leaving the equipment cold-plates, through the heat exchanger (B-7). The bypass valve (B-8) continues to control the cabin temperature. When danger of low cabin temperature has passed, the warm glycol recirculation is terminated by manually closing valve B-12.

A spring loaded glycol accumulator (C-2) provides a constant pressure of 40 psia at the glycol pump inlet. The pressure of 40 psia was selected as the minimum glycol pressure which will preclude vapor formation in the loop. The accumulator has an automatic low-level shut-off valve (C-3) which shuts off accumulator make-up flow when the accumulator level drops below 50%. This shut-off function is provided to prevent loss of the entire glycol supply in the event of a failure of the radiator or the occurrence of a repairable leak in the glycol circuit.

Three parallel positive displacement glycol pumps (C-5), each with a check valve (C-6) to prevent reverse flow, provide flow through the glycol loop. During lunar orbit, descent and rendezvous when the electrical heat loads are high, two of the pumps are operated to provide a flow rate of approximately 127 lb/hr. at a differential pressure of approximately 20 psi. The third pump is normally off and provides redundancy should one of the other two pumps fail. During the lunar stay, the electrical loads are reduced considerably and a lower flow rate can be utilized. During this period, only one pump is operated which provides a glycol flow rate of approximately 52 lb/hr. Since the glycol pumps are of the positive displacement type, excessive pressures would result if the glycol line became clogged. A pressure relief valve (C-4) is provided to relieve excessive pressures and prevent rupture of the lines. Clogging of the lines would cause abort of the mission and would require reliance on the emergency equipment cooling system.

Essential electrical and electronic equipment is cooled by passing the glycol through cold plates (C-7) to which the equipment is mounted. Glycol is passed through the cold plates in series, cooling the equipment with low temperature limits (such as the guidance package) first and then progressing to the equipment with higher temperature limits.

The warm glycol passes through the cabin wall to the fuel cell cryogenic storage heat exchangers. A relatively small amount of heat is removed from the glycol by eight cryogenic heat exchangers which are used for pressurization of the electrical power fuel supply. After the glycol has passed through the radiator and its controls, it passes through the two cryogenic heat exchangers (E-2) for pressurization of the environmental oxygen tank. The environmental oxygen pressurization heat load has a negligible effect on the glycol temperature.

Heat is dissipated from the glycol by a fin-tube space radiator (C-14). The radiator is built as a separate unit rather than being integrated with the structure. Integration with the structure presents difficulty because of the large temperature gradients required for thermal control. In addition, a separate radiator may be varied in size according to mission design requirements much more easily than an integrated radiator and imposes only a small weight penalty. Lunar hot side operation requires a large radiator which is oriented parallel to the lunar surface. The latter requirement cannot be accomplished with an integrated radiator on the recommended vehicle.

Control of the heat dissipated by the radiator is provided by a regenerative glycol heat exchanger (C-12) and one of the two redundant bypass control valves (C-11). The radiator is sized to provide sufficient cooling for average lunar rest heat loads. For peak load conditions, the control valve (C-11) senses the temperature of the glycol downstream of the control loop. As the radiator outlet temperature drops below 40°F., part of the flow is bypassed through the regenerative heat exchanger where it is heated by the hot glycol entering the radiator. Sufficient flow is bypassed to maintain a downstream glycol temperature of 40°F. A selector valve (C-10) diverts the glycol flow to one of two control valves. The two check valves (C-6) prevent back flow through the control valve not being used.

The water boiler (C-1) is used to augment the radiator cooling. Water is stored in the water coolant tank (D-4) for both crew consumption and cooling. Water from the storage tank is supplied to the water boiler by one of two redundant control valves (D-3). The control valve controls the flow of water to the boiler to maintain the glycol temperature downstream of the boiler at 40°F. Manual selector valves (D-1) are positioned to select the desired control valve of operation.

Prior to lunar launch the radiator and the cryogenic tankage is off-loaded on the lunar surface. This breaks the glycol lines outside the pressurized cabin. In order to maintain glycol circulation, the bypass valve (C-9) is actuated and the entire glycol system outside the spacecraft cabin is bypassed. The water boiler then provides the entire cooling function for the launch and rendezvous phase of the mission. The external portion of the glycol loop may also be bypassed in the event of failure of the radiator or any of the external components.

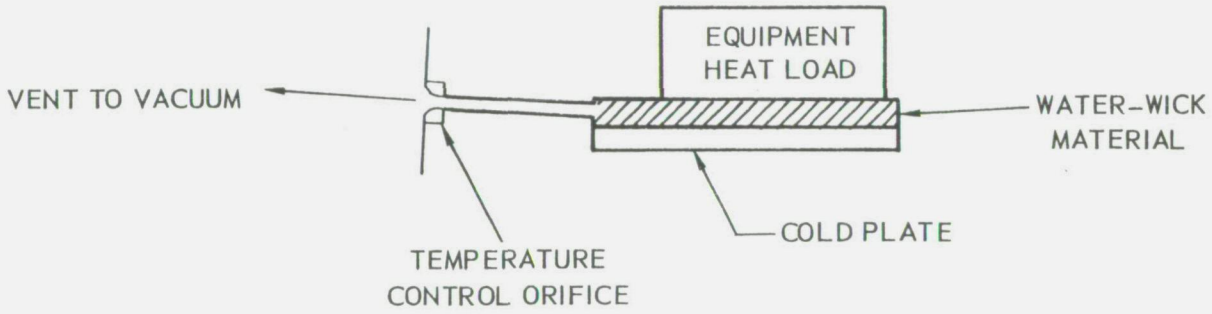
16. 3. 5. 2. 4 Cold Plate Water-Wick Boiler Operation

Water-wick type boilers are built into each equipment cold plate to provide emergency cooling to the electrical equipment in the event of a failure of the glycol loop (Figure 16-40). During translunar flight and during normal thermal control system operation, the water boilers are sealed so that the water does not vaporize prematurely. The boiler vent lines tie into a common header. Emergency cooling is initiated by opening a valve in the header line. The boiler temperature is controlled by an orifice in the header vent. The temperature at which the water evaporates is self-regulated. As the temperature of the boiler drops, the water vapor pressure is reduced and the specific volume of the water vapor increases. The mass flow of water through the orifice also decreases which stabilizes the temperature. Figure 16-45 shows the variation of boiler temperature as a function of the rate at which heat is absorbed. This curve is based on a boiler temperature of 70°F at 100% design heat load.

If analyses of the detailed design of the cold plates indicate that the minimum temperature required falls below the water freezing point, the temperature capabilities can be lowered by additives to the water. The freezing point can be lowered and the range of cooling increased at the expense of increased coolant weight. Figure 16-46 shows the variation of freezing point and the weight of coolant required per 1, 000 BTU dissipated as a function of the percent of ethylene glycol added to the water. Based on preliminary analysis of the cold-plates, however, it is not anticipated that additives will be required.

16. 3. 5. 2. 5 Water Management

Water stored before launch and the water produced by the fuel cells is used for drinking, food preparation and cooling. The fuel cells have an accumulator capable of storing the water produced during a period of 24 hours. The water produced in the fuel cells is stored at approximately 140°F and 60 psia. This water is transferred periodically to the coolant water tank (D-4) by opening a shut-off valve (D-1) in the connecting line. The higher pressure in the fuel cell accumulator forces the water into the



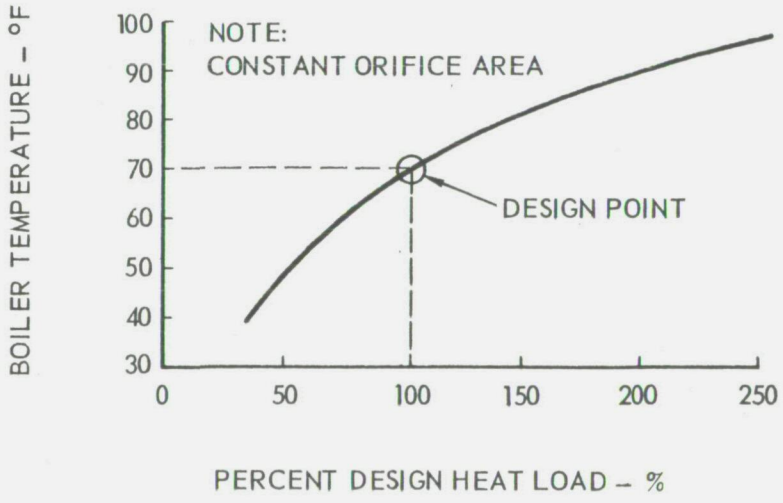


Figure 16–45 WATER WICK BOILER PERFORMANCE

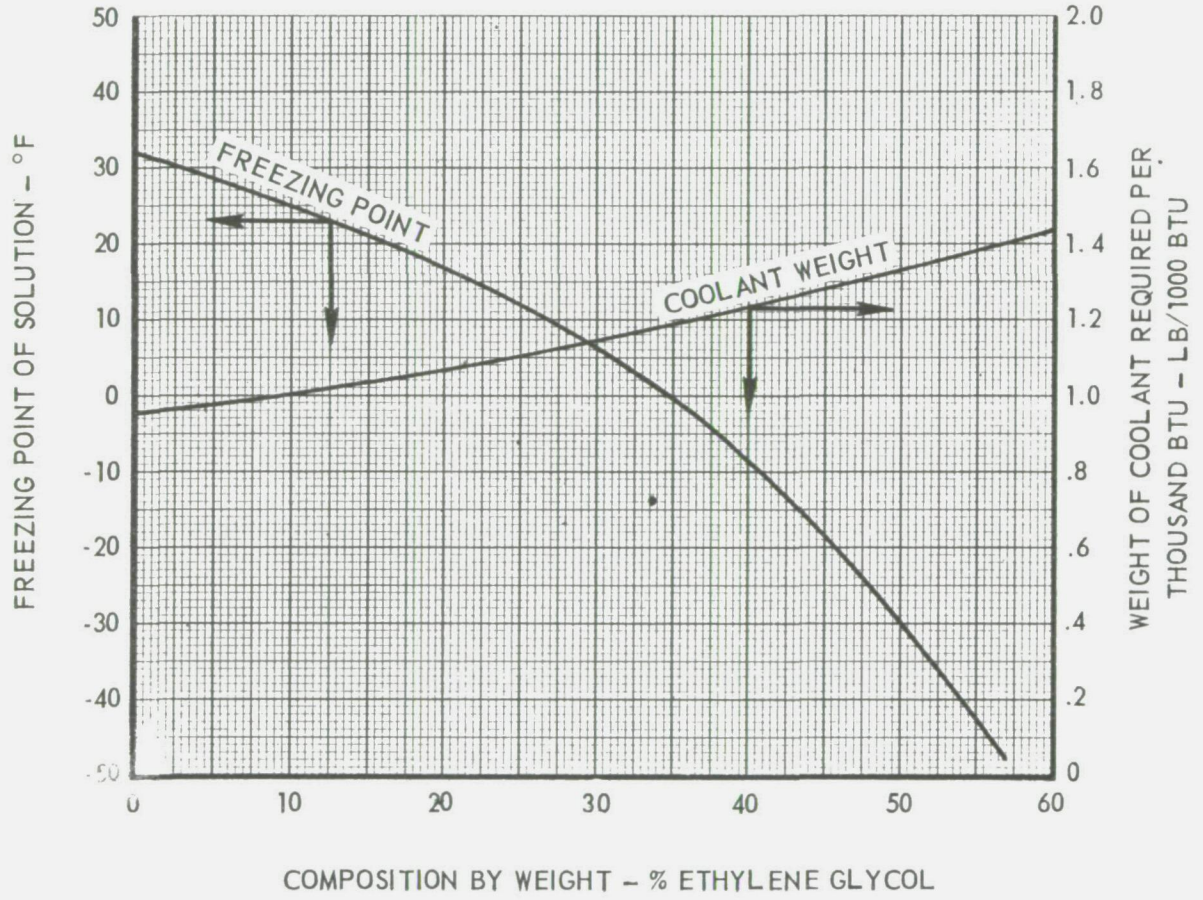


Figure 16-46 FREEZING POINT AND COOLANT WEIGHT FOR WATER - ETHYLENE GLYCOL COOLANT SOLUTIONS

coolant tank. Water condensed and separated from the atmosphere is also stored in the coolant water tank. The pressure head built up by the centrifugal separators (A-20) is sufficient to flow the water into the coolant tank.

Urine and waste wash water are stored in a separate tank (D-6) during the mission. Before storage in the tank, the waste water is sterilized by passage through an ion-exchange filter. The waste water is brought back to lunar orbit and transferred to the command module for transport back to earth. This is consistent with the Apollo philosophy of bringing all waste water back to earth to prevent possible contamination of the lunar surface. Should this policy be rescinded, the waste water tank will be sealed and left on the lunar surface prior to lunar launch. This would provide a significant weight savings, especially for a 160 hour mission. (Paragraph 16.3.2.10)

16.3.5.3 Power Summary

The electrical power requirements for the recommended environmental control system are summarized in Table 16-XIII. This table shows the power requirements for: (1) the normal operation in lunar orbit (after the environmental control system has been activated), (2) lunar landing and rendezvous, (3) normal operation during lunar surface operations, and (4) emergency operation. The cabin fan is not operated during the descent and rendezvous maneuvers since the LEM cabin is unpressurized. During these maneuvers, two glycol pumps are operated to provide a high glycol flow rate. During lunar surface operations, only one pump is used. It is not anticipated that the catalytic burner will be used during descent or ascent of the LEM. It is required only during extended lunar surface operations. The power requirements listed for emergency operation are for an emergency abort in which the power has been reduced to a minimum.

16.3.5.4 Weight Justification

History shows that the weight of new components and systems tends to increase from the initial estimates during development. Many of the components in the recommended ECS are modifications of existing items. Therefore, a survey of similar equipment was made in order to provide weight estimates representative of actual equipment rather than ideal numbers. The results of this survey are described in the following paragraphs. The system weight breakdown is shown in Table XII.

(a) Water and water tankage (73.3 lbs.) - The water and water tankage include the following items:

	Fluid Weight	Tankage Weight
(1) Waste water tank - capacity 16.5 lbs.	----	5.1
(2) Cooling water tank	53.3	8.8

The weight of drinking and cooling water consumed is determined by the heat which must be dissipated by the water boiler during transient peak heat load conditions and by the metabolic and wash water

TABLE 16 –XIII ECS POWER REQUIREMENTS SUMMARY

NO.	QTY.	ITEM	NORMAL, LUNAR ORBIT, LANDING AND RENDEZVOUS		NORMAL LUNAR SURFACE OPERATIONS		EMERGENCY	
			WATTS CONT.	WATTS INT.	WATTS CONT.	WATTS INT.	WATTS CONT.	WATTS INT.
1	1	SUIT COMPRESSOR	166	–	166	–	–	–
2	1	CABIN FAN	–	–	66	–	–	–
3	1	OXYGEN FLOW REGULATOR	–	4	–	4	–	–
4	2	GLYCOL BYPASS VALVE	–	20	–	20	–	–
5	1	H_2O FLOW CONTROL VALVE	–	7	–	7	–	–
6	1	CATALYTIC BURNER	–	–	–	75*	–	–
7	1–2	GLYCOL PUMP	60	–	30	–	–	–
8	1	CONTROL PANEL	24	–	24	–	24	–
9	1	EMERGENCY COOLING VALVE	–	–	–	–	7	–
		TOTALS	250	31	286	106	31	–

* USED ONLY AS NEEDED

NOTE: WATTS ARE DC OR DC EQUIVALENT

requirements. The water generated by the fuel cells is utilized in meeting part of these requirements. The transient heat loads which require water boiler augmentation of the radiator occur during checkout and landing with short periodic increases in vehicle power usage during operation on the lunar surface. The water boiler provides the entire heat dissipation during lunar launch and rendezvous since the radiator is off-loaded. The cooling water required has been calculated on the basis of the integrated values of these heat loads for the complete mission. The equation used to calculate coolant weight is as follows:

$$W_w = \frac{Q_t}{h_{f_g}}$$

where: W_w = weight of cooling water required - lbs.

Q_t = integrated water boiler heat loads over the complete mission (i.e., heat loads above the capacity of the vehicle radiator) - BTU

h_{f_g} = heat of vaporization of water at the design boiling temperature - $\frac{BTU}{lb.}$

Waste water tankage capacity is a function of the amount of urine produced by the crew and the wash water used. Man's average requirements presented by Table 16-II were used as a basis for determining the tank capacity required.

Figure 16-47 shows the comparative data used to establish the water tank weights for the LEM. Tank weight is plotted as a function of fluid weight from estimated vendor data on other systems. The actual tank weight for the Mercury system is shown on the curve and is significantly higher than the estimated values. The LEM water tank weights were chosen as an average between installed Mercury weights and the vendor estimates. This average was used since it appears that the Mercury weight can be lowered but probably not to the degree indicated by the vendor estimates.

(b) Oxygen and Oxygen Tankage (92.9 lbs.) - The oxygen and oxygen tankage weight include the following:

(1) Normal 0_2 supply	50.0 lbs.
Metabolic requirements 2 lbs/man-day	4.19
Leakage at .1 lb/hr.	2.51
Two cabin repressurizations 11.2 lbs. each	22.40
Four airlock operations at 1.05 lbs. each	4.20
50% excess	16.70

Note: Refills for backpacks to be taken from the 50% excess and from the normal metabolic supply.

(2) Normal 0_2 tankage 30.0 lbs.

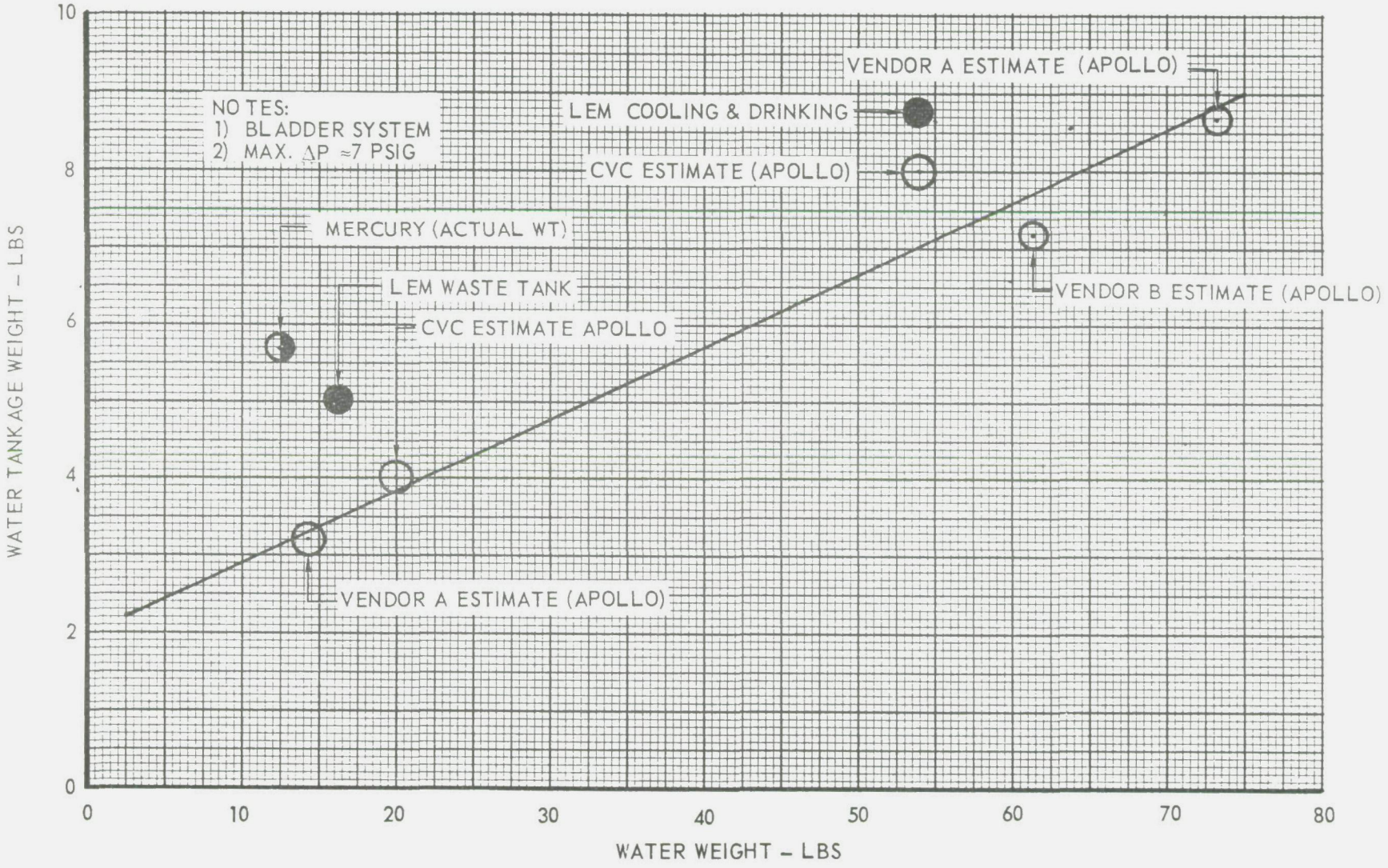


Figure 16–47 WEIGHT OF SPACE VEHICLE WATER TANKAGE

(3) Rendezvous oxygen	4.0 lbs.
Leakage at .1 lb/hr.	0.4
Metabolic requirements at 2.0 lb/man-day	0.6
One backpack refill (estimate)	2.0
Reserve	1.0
(4) Rendezvous oxygen tank	8.9 lbs.

The 2 lbs/man-day metabolic 0_2 consumption is the accepted average daily metabolic requirement. The assumed leakage rate of .1 lb/hr. was established using extrapolated Project Mercury data. Two cabin repressurizations are provided for both the command module and the LEM. For the LEM, one cabin repressurization is required for the initial filling and one is required as a contingency. The four airlock operations are based on the expected entrance and exit requirements. The current command module specification calls of 18 airlock operations for a 7-day period on the lunar surface. The 50% excess is the same as the current command module specification requirements and includes provisions for excessive leakage, backpack refills, boil off, etc.

The backpack refill included in the rendezvous oxygen is a contingency since the backpacks should normally be filled before leaving the lunar surface. The rendezvous oxygen weight and tank are identical to one Mercury oxygen tank. Vendor weight estimates for this size of gaseous oxygen tank agree very closely with the Mercury weight.

Figure 16-48 shows the data used to establish the LEM ECS cryogenic 0_2 tankage assembly weights. The data presented in Reference (2) has been used as the basis for the LEM weights. The vendor estimates for Apollo were considerably lighter than the Reference (2) data due to the fact that titanium tankage was proposed. Since that time a problem of materials compatability has arisen and titanium tanks are not recommended for cryogenic oxygen storage.

(c) CO_2 and Odor Removal System (LiOH and Activated Charcoal) (16.9 lbs.)

For the LEM 1-day mission this system consists of two canisters containing sufficient LiOH and activated charcoal for removal of the CO_2 and odors generated. Figure 16-49 shows a comparison of the LEM weights with available data on actual and calculated weights. Also shown on Figure 16-49 is a curve representing the ideal weight of LiOH required with 100% absorptive efficiency. The weight between these two curves represents the weight of the charcoal, canister, filters, etc.

For extended LEM missions, a 1-day canister will be provided for each day of the mission. The two canister housings provide a means of interchanging fresh canisters for expended ones without disrupting 0_2 flow.

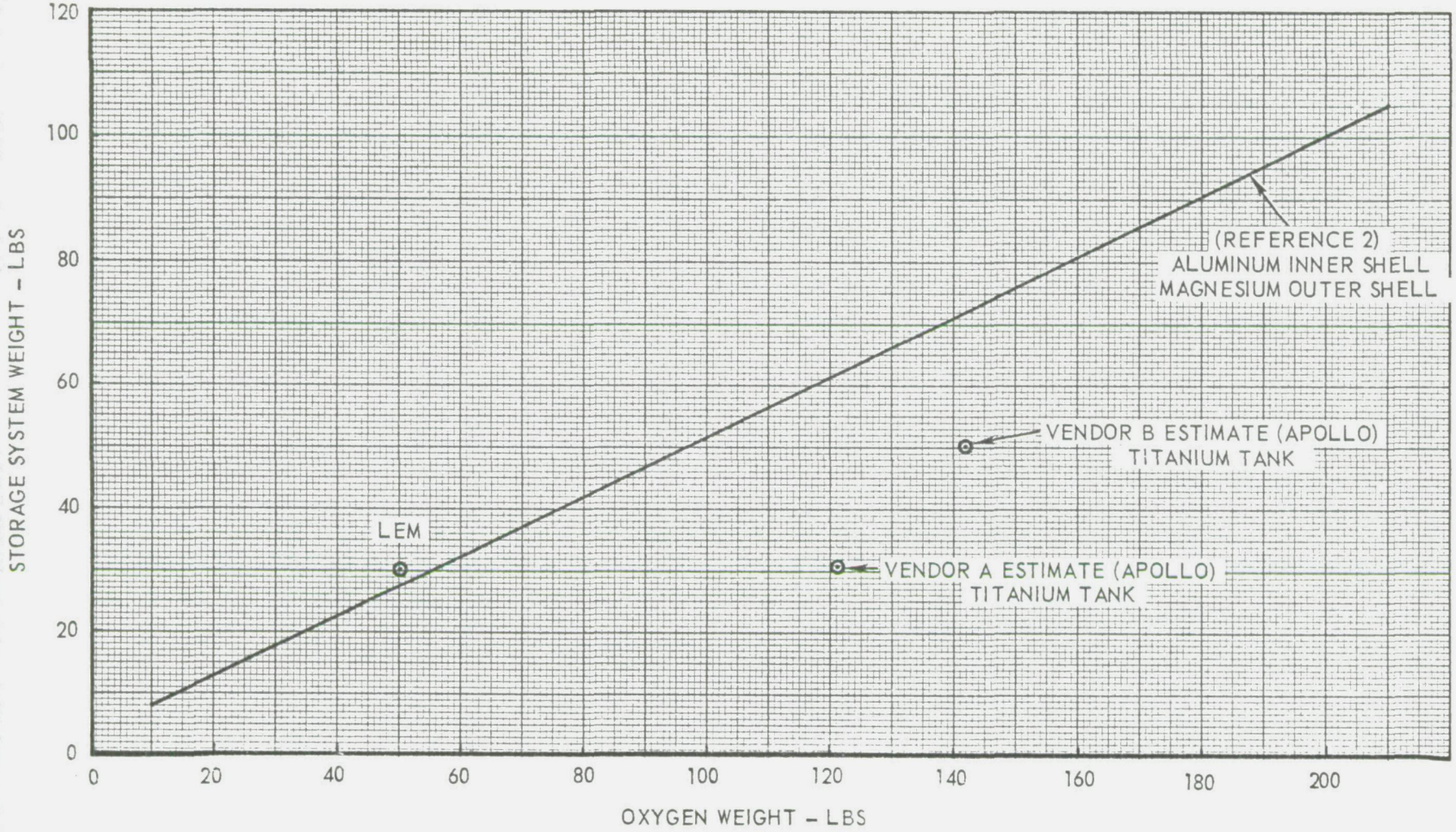


Figure 16-48 WEIGHT OF SUPERCRITICAL OXYGEN STORAGE SYSTEMS

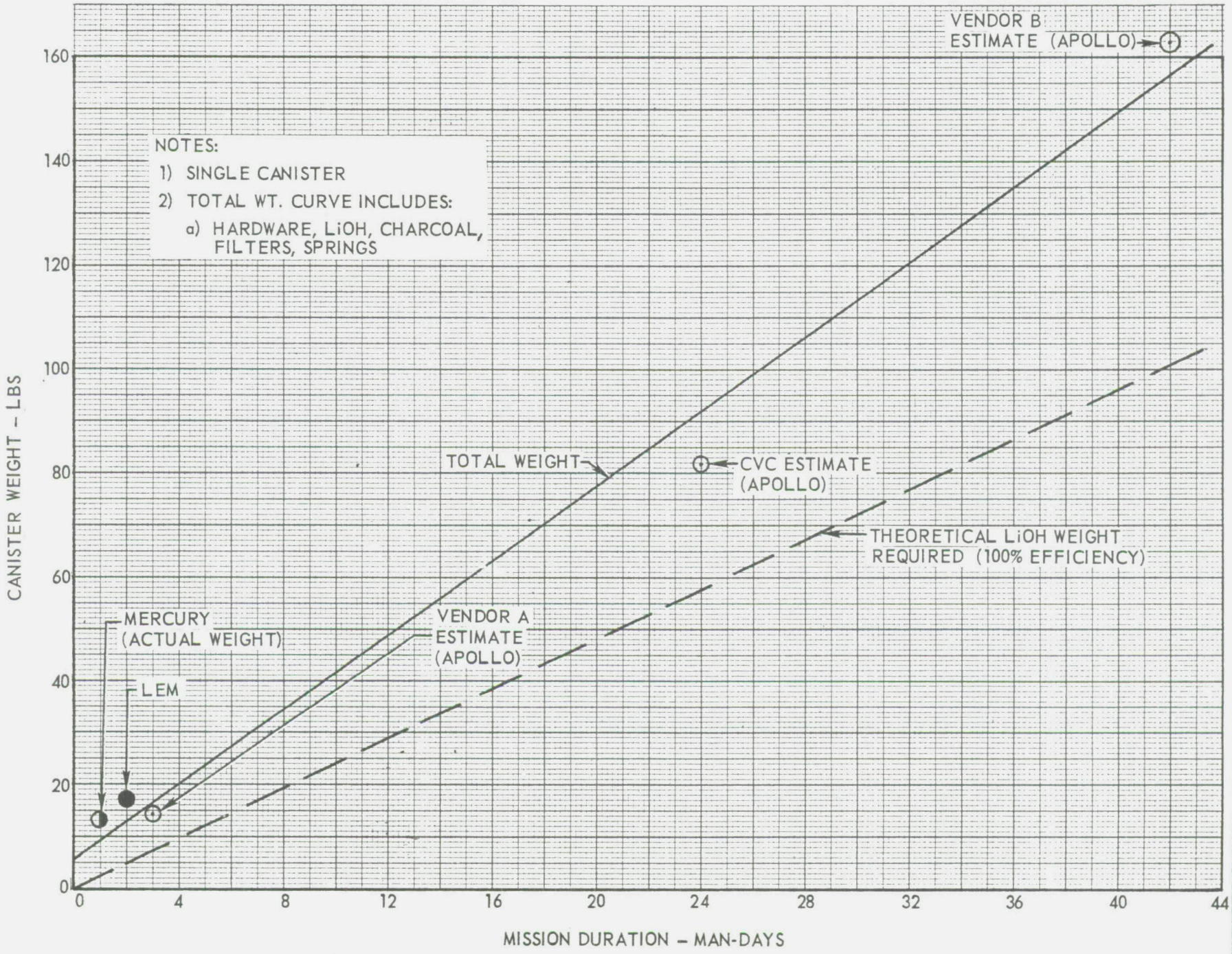


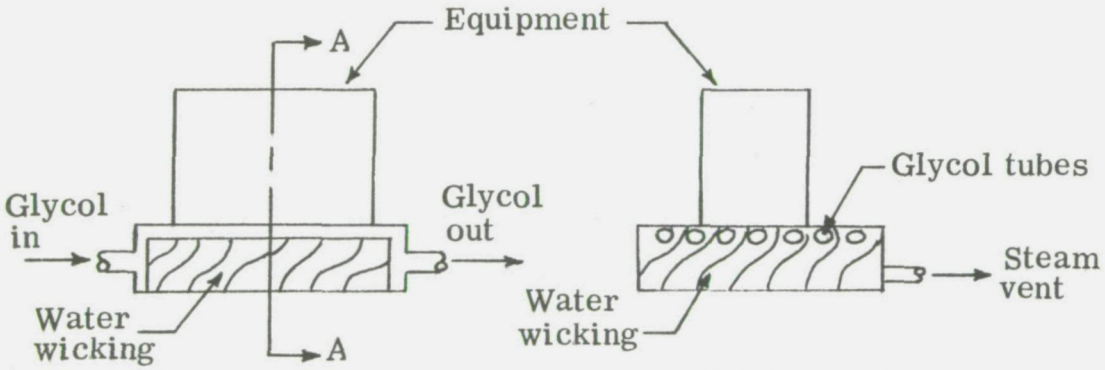
Figure 16–49 WEIGHT OF LITHIUM HYDROXIDE AND CHARCOAL CANISTERS

(d) ECS Radiator - 31.5 lbs. (cold side operations only)

The ECS radiator size for the LEM is based on the integrated average heat load while at rest on the lunar surface. As noted in an earlier paragraph, cooling water is used for transient peak heat load conditions. Based on the current electrical heat loads and a maximum equipment cold plate temperature of $160^{o}F$, the radiator area required is 35 ft^2. Calculations show that non-integrated glycol radiators operating in this temperature range, and their supporting structure, weigh approximately .9 lbs/ft^2. This is the basis for the selected radiator weight.

(e) Equipment Cold Plates - 64.6 lbs.

An accurate weight estimate for the LEM equipment cold plates is difficult since the proposed design is unique. The LEM cold plates will contain water and water wicking for emergency cooling and glycol flow passages for normal cooling. (See sketch below)



Section A-A

As shown by the sketch, the construction of the cold plates is closely akin to the construction of a shell and tube heat exchanger. Shell and tube heat exchanger core weights which have been obtained in compact aircraft type heat exchangers are approximately 34.1 lbs/ft^3 for aluminum construction and 62.4 lbs/ft^3 for steel construction (Reference 2).

The emergency cooling water weight is determined by the heat loads and time required for abort. Using current heat loads and a 4-hour time requirement, the water weight is 34.3 lbs. Using the above quoted values for heat exchanger core weight for aluminum construction, the core weight for 10 ft^2 of cold plates 3/4 inches thick is 21.4 lbs. (Core volume = .626 ft^3)

This weight is for the basic core only, and does not include manifolds, header plates, and supporting structure. Studies of weights of production heat exchangers for aircraft application indicates that the total weight may be related to the core weight by the following equation:

$$\frac{W_{total}}{W_{core}} = \left(\frac{12}{V}\right).118$$

where: V = core volume in ft^3

Using this equation the estimated total equipment weight for the cold plate is 30.3 lbs. This value has been used for the LEM cold plate weight.

(f) Miscellaneous Ducting, Plumbing, and Other Installation Hardware - 86.7 lbs.

An accurate weight estimate for these miscellaneous items is the most difficult of all to obtain. The Mercury program provides the only available data on existing manned spacecraft on which to base LEM installation weights. The breakdown on the Mercury environment control system is as follows:

	% Total Weight
(1) Major hardware - valves, heat exchangers, storage tanks, compressors, etc. (dry)	50
(2) Expendable materials, i.e., H_2O, O_2, LiOH	21.8
(3) Mechanical components for installation, i.e., brackets, ducts, lines, seals, clamps	19.4
(4) Miscellaneous electrical installation equipment (controls, cables, sensors, indicators, switches, etc.)	8.8
	100.0%

The percentage of mechanical and electrical installation equipment weight can be expected to decrease with an increasing overall system weight, however, no experience is currently available to indicate how much this decrease will be. Based on the above data, a value of 20% was selected for the LEM. This allows 15% for mechanical equipment and 5% for electrical equipment.

The remaining environmental control system components, such as lightweight valves, compressors, separators, pumps, etc., account for approximately 20% of the total weight. For these items weights have been obtained from vendor estimates on the Apollo Command and Service Modules and from actual hardware weights on Project Mercury.

16.3.5.5 Changes For a 7-Day Operation (Lunar Cold Side)

Extension of mission duration to seven days on the lunar cold side requires additional quantities of the following ECS expendable components.

(a) H_2O and H_2O supply tanks
(b) O_2 and O_2 supply tanks
(c) Waste H_2O tank
(d) LiOH and canisters (including charcoal)

The time-variant weights of these items as shown in Figure 16-50 as a function of stay time on the lunar surface.

The initially reversed slope of H_2O and H_2O tankage weight curve results from the fuel cell H_2O generation rate exceeding the total H_2O usage rate during lunar rest operation. This excess water can be stored and used for rendezvous cooling, allowing a corresponding reduction in the quantity of water which must be stored prior to earth launch. A quantity of excess water equal to the total rendezvous requirements is accumulated in approximately 50 hours, after which excess water is stored in mylar bags and off-loaded on the lunar surface.

The lithium hydroxide CO_2 removal system weight is shown to increase in steps due to the packaging of lithium hydroxide and activated charcoal in 24-hour canisters.

Metabolic requirements, cabin leakage, airlock operations (4 per day), and a 50% contingency determine the O_2 and O_2 tankage weight increase for the longer mission. The slope of the O_2 and O_2 tankage weight curve, as well as that for waste H_2O tankage weight is relatively independent of mission duration.

The ECS weight penalty incurred by extending the mission from 24 to 160 hours on the lunar cold side may be obtained by summing the time-variant weights of the above components. This summation results in a weight penalty of 166 lbs. for a total ECS weight of 629 lbs. for a 160-hour lunar stay.

16.3.5.6 Changes for Lunar Hot Side Operation

As discussed in Paragraph 16.3.4.5 (Structural Heat Leakage), lunar hot side landings increase the heat absorbed through the cabin walls at a time when rejection of heat is difficult. This additional heat leakage increases the coolant weight required for ECS operation. The minimum of 0.10 inch of "super-insulation" provided for cold side lunar landings results in optimum system weight for hot side lunar stays up to about 80 hours. Since changing to an optimum insulation thickness for longer missions will not affect a large weight savings, the insulation thickness of 0.10 inch is used for all missions. This is also in keeping with the concept of minimum changes in the LEM for extended mission capability.

One of the major problems associated with lunar hot side landings is the window. The window is heated by radiation from the lunar surface in addition to solar radiation. Excessive heat loads on the window can best be eliminated by covering the exterior of the window with a low emissivity radiation shield. For lunar hot side landings, external shutters will be provided that can be positioned manually from within the LEM. Should difficulty be encountered in positioning the shutters, they can be adjusted by a crewman working outside the LEM.

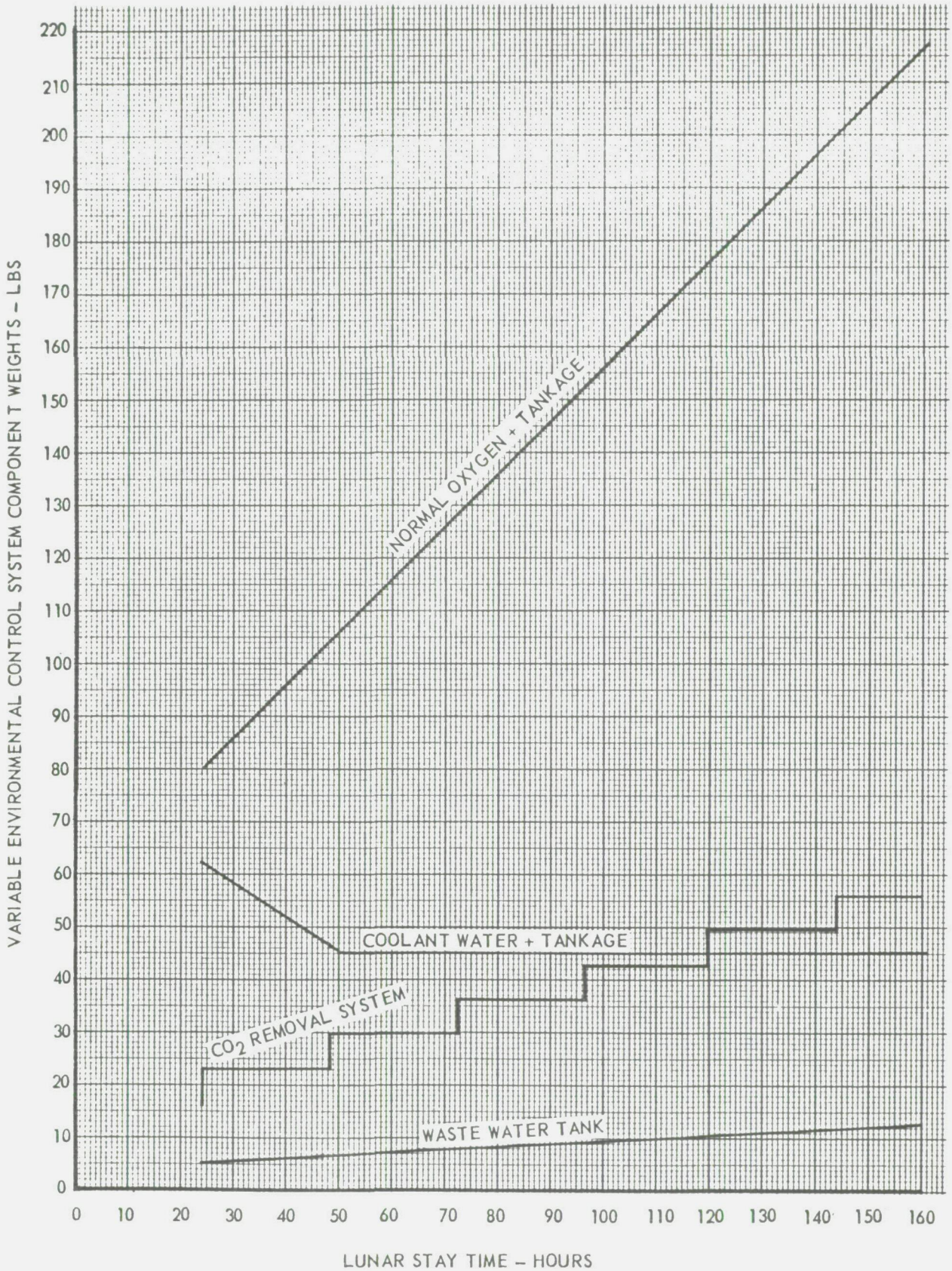


Figure 16–50 EFFECT OF LUNAR COLD SIDE STAY TIME ON ENVIRONMENTAL CONTROL SYSTEM COMPONENTS WEIGHT

The changes required for a lunar hot side landing as compared to a cold side landing are shown in Table 16-XIV. The radiator and coolant weights were determined assuming the radiator would be off-loaded on the lunar surface. Off-loading the radiator changes the optimum radiator area and the weight penalty associated with the cooling system because no lunar launch propulsion is required. The figure indicates an environmental control system weight penalty for a 24-hour lunar hot side landing of 124 lbs. and 318 lbs. for a 160-hour lunar hot side stay.

16.3.5.7 Controls and Displays

Sufficient controls and displays are provided for complete in-flight checkout of the environmental control system and for monitoring the essential functions throughout the mission. Warning lights are actuated by the improper operation of any major component or subsystem. A summary of the warning lights is presented in Table 16-XV. Table 16-XVI presents a summary of the indicators. All quantities required to determine the complete operation of the ECS can be monitored on the control panel. The phase of flight in which the data is displayed and transmitted back to earth are also shown. The controls required for ECS operation are summarized in Table 16-XVII. Where continuous control or immediate response to an error signal is required, automatic controls were used. Back-up modes to the automatic systems and control functions that only require infrequent attention are provided by manual operation since the manual systems are basically more reliable.

16.3.5.8 Ground Checkout Requirements

Operation of the environmental control system will be thoroughly tested prior to flight. The service, test, and checkout of the subsystems and components will require special ground support equipment. A summary of the major tests and checkout items during field hanger test and checkout and earth prelaunch test and checkout is shown in Table 16-XVIII. Fill and service capability will be required for the oxygen supply, glycol system, and water coolant systems during this time. A freon ground cooling system provides cooling for the equipment during checkout. The controls and displays provided with the environmental control system are sufficient for in-flight test and checkout. These controls and displays are described in Section 16.3.5.7.

TABLE 16 – XIV ECS CHANGES FOR LUNAR HOT SIDE LANDING

	LUNAR STAY TIME					
	24 HR.		80 HR.		160 HR.	
	HOT SIDE	COLD SIDE	HOT SIDE	COLD SIDE	HOT SIDE	COLD SIDE
CABIN WALL INSULATION (EXTERIOR)	0.10 INCH SUPER – INSULATION →					→
WINDOWS (WITH LOW EMISSIVITY COATING)	EXTERNAL SHUTTERS	INSIDE CURTAIN ONLY	EXTERNAL SHUTTERS	INSIDE CURTAIN ONLY	EXTERNAL SHUTTERS	INSIDE CURTAIN ONLY
ELECTRICAL EQUIPMENT	WELL INSULATED	MINIMUM INSULATION	WELL INSULATED	MINIMUM INSULATION	WELL INSULATED	MINIMUM INSULATION
RADIATOR AREA FT^2	53	35	158	35	230	35
COOLANT WATER AND TANKAGE LBS.	122	62.5	220	45.3	232	45.3

TABLE 16-XV ECS WARNING LIGHTS

NO.	WARNING LIGHT	CONDITION CAUSING SIGNAL FROM WARNING LIGHT
1.	SUIT COMPRESSOR	OVERTEMP OF EITHER SUIT COMPRESSOR MOTOR OR A ΔP ACROSS COMPRESSOR LESS THAN 13" H_2O
2.	CO_2 PARTIAL PRESSURE	PARTIAL PRESSURE OF CO_2 >15.2MM HG IN SUIT CIRCUIT
3.	RADIATOR OUTLET TEMP.	RADIATOR OUTLET TEMPERATURE < -40°F
4.	GLYCOL PUMP	OVERTEMP OF ANY GLYCOL PUMP MOTOR OR BY A LOW OR HIGH ΔP ACROSS GLYCOL PUMPS
5.	GLYCOL ACCUMULATOR	LOW LEVEL IN GLYCOL ACCUMULATOR
6.	CABIN AND SUIT PRESSURE	CABIN PRESSURE OR SUIT PRESSURE < 4.5 PSIA,
7.	STEAM VENT	H_2O BOILER VENT STEAM TEMPERATURE <38°F
8.	O_2 SUPPLY	O_2 PRESSURE IN SUPPLY LINE <800 PSIA
9.	CABIN FAN	OVERTEMP. OF EITHER FAN MOTOR OR LOW FAN ΔP
10.	COOLING WATER	LOW LEVEL IN STORAGE TANK OR LOSS OF PRESSURIZATION
11.	REND. O_2 FLOW	USAGE OF RENDEZVOUS OXYGEN
12.	RELATIVE HUMIDITY	RELATIVE HUMIDITY OVER 60%

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

TABLE 16-XVI ECS INDICATORS AND DISPLAYS

ITEM NO.	DATA	RANGE	MODE OF READ-OUT								REMARKS
			LUNAR ORBIT		LUNAR DESCENT		LUNAR REST		RENDEZVOUS		
			T	D	T	D	T	D	T	D	
1	CO_2 PARTIAL PRESSURE	0-20 MM Hg	X	X	X	X	X	X	X	X	
2	CABIN TOTAL PRESS.	0-300 MM Hg		X		X		X		X	
3	SUIT TOTAL PRESS.	0-300 MM Hg	X	X	X	X	X	X	X	X	
4	CRYOGENIC O_2 TANK PRESSURE	0-1000 PSIA	X	X	X	X	X	X	X	X	
5	CRYOGENIC O_2 TANK TEMP.	-350 TO 100°F		X		X		X		X	
6	GLYCOL FLUID RESERVE	0-FULL	X	X	X	X	X	X	X	X	
7	GLYCOL TEMP.										1 INDICATOR SWITCHING ARRANGEMENT (5 POSITION)
	a. REGEN HX INLET	-60 TO 250°F		X		X		X			
	b. RADIATOR OUTLET	-60 TO 250°F	X	X	X	X	X	X			
	c. H_2O BOILER OUTLET	-60 TO 250°F	X	X	X	X	X	X	X	X	
	d. GUIDANCE PACKAGE OUTLET	10 TO 250°F	X	X	X	X	X	X	X	X	
	e. REGEN. HEAT EXCHANGER OUTLET	-60 TO 250°F		X		X		X			
8	SUIT CIR. REL. HUMIDITY	0 – 100%	X	X	X	X	X	X	X	X	
9	CABIN TEMP.	0 – 200°F	X	X	X	X	X	X	X	X	
10	COOLING WATER TANK LEVEL	0, – 100%	X	X	X	X	X	X	X	X	
11	GAS ANALYZER	O_2, CO_2, H_2S, H_2, CH_4, CO						X			
12	SUIT AIR TEMP.	0 – 200°F	X	X	X	X	X	X	X	X	
13	O_2 QUANTITY	0 – FULL	X	X	X	X	X	X	X	X	
14	AIRLOCK PRESSURE	0 – 15 PSIA									
15	REND. O_2 PRESS	0 – 7500 PSIA									
16	O_2 FLOW RATE	0 – 1 LB/MIN.	X	X	X	X	X	X	X	X	

*T – TRANSMITTED AND RECORDED
D – DISPLAYED ON CONTROL PANEL

TABLE 16-XVII ECS CONTROLS SUMMARY

ITEM	FUNCTION	TYPE	REMARKS
1	SUIT COMPRESSOR SELECTOR	3 POSITION SWITCH	OFF, FAN 1, FAN 2
2	CABIN FAN SELECTOR	3 POSITION SWITCH	OFF, FAN 1, FAN 2
3	GLYCOL PUMP SELECTOR	3 ON-OFF SWITCHES	PUMP 1, PUMP 2, PUMP 3
4	CABIN TEMP. CONTROL	VARIABLE CONTROL	
5	CABIN DECOMPRESSION AND REPRESSURIZATION	GUARDED 3 POSITION SWITCH	REPRESS-OFF-CABIN DUMP
6	O_2 TANK PRESSURIZATION	ON-OFF SWITCH	FOR O_2 TANK HEATER
7	RADIATOR LOOP SELECTOR	3 POSITION SWITCH	LOOP 1, LOOP 2, BYPASS
8	CATALYTIC BURNER	ON-OFF SWITCH	ACTIVATE HEATER
9	O_2 NORMAL CONTROL	2 POSITION SWITCH	OFF-NORMAL OXYGEN
10	GLYCOL REGENERATOR MODULATING VALVE	2 POSITION SWITCH	VALVE 1, VALVE 2
11	EMERG. COOLING	ON-OFF SWITCH	OFF-EMERG. COOLING
12	NORM. OXY. SUIT PRESS	3 POSITION MANUAL	OFF, VALVE 1, VALVE 2
13	CRYO. PRESS. VALVE SELECTOR	5 SWITCHES, 2 POSITION	VALVE 1, VALVE 2

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

TABLE 16 – XVIII
GROUND CHECKOUT REQUIREMENTS

CABIN TESTS RELIEF VALVE LEAK CHECK FAN OPERATION
O_2 SUPPLY, SUPERCRITICAL AND GASEOUS PRESSURIZATION SYSTEM REGULATORS (ALL) LEAK CHECK
SUIT CIRCUIT H_2O SEPARATOR OPERATION COMPRESSOR OPERATION CATALYTIC BURNER FLOW RELIEF VALVE LEAK CHECK
GLYCOL SYSTEM SUIT TEMP. CONTROLS CABIN TEMP. CONTROLS RADIATOR AND BY PASS CONTROLS GLYCOL PUMP OPERATION FLOW RATE AND TEMPERATURES LEAK CHECK
H_2O SYSTEM H_2O BOILER CONTROLS LEAK CHECK

DEVELOPMENT PROGRAM

16.4 DEVELOPMENT PROGRAMS

Figure 16-51 shows a development schedule for the Environmental Control system which is based on the over-all LEM Program schedule described in Volume VIII Section 1.0. The ECS schedule shown by Figure 16-51 appears realistic considering the current development status of ECS components selected for the LEM.

Some of the major environmental system tasks associated with development of the selected ECS system and vehicle configuration are as follows:

(a) Development and proof testing of a good low emissivity - high visible transmissivity thermal control coating for the large window area.

(b) Determination of a suitable installation technique for the structural insulation.

(c) Analysis and proof testing of passive thermal control techniques for the material and equipment outside the vehicle pressure shell.

(d) Development and testing of a low leakage seal for the connection between the LEM and the Apollo command module which will permit an "open door" operation during the translunar phase. (Note - airlock doors may be closed if a suitable seal cannot be developed).

(e) Integration studies to prove out the selected method of combining the command module and LEM environmental control system functions during the translunar phase of the mission.

Some work has already been accomplished on all of the above tasks and there are no obvious areas which would cause lack of sufficient time to meet the proposed schedule. The ECS equipment components can to a large degree be scaled, and in some cases are actually available from current development programs.

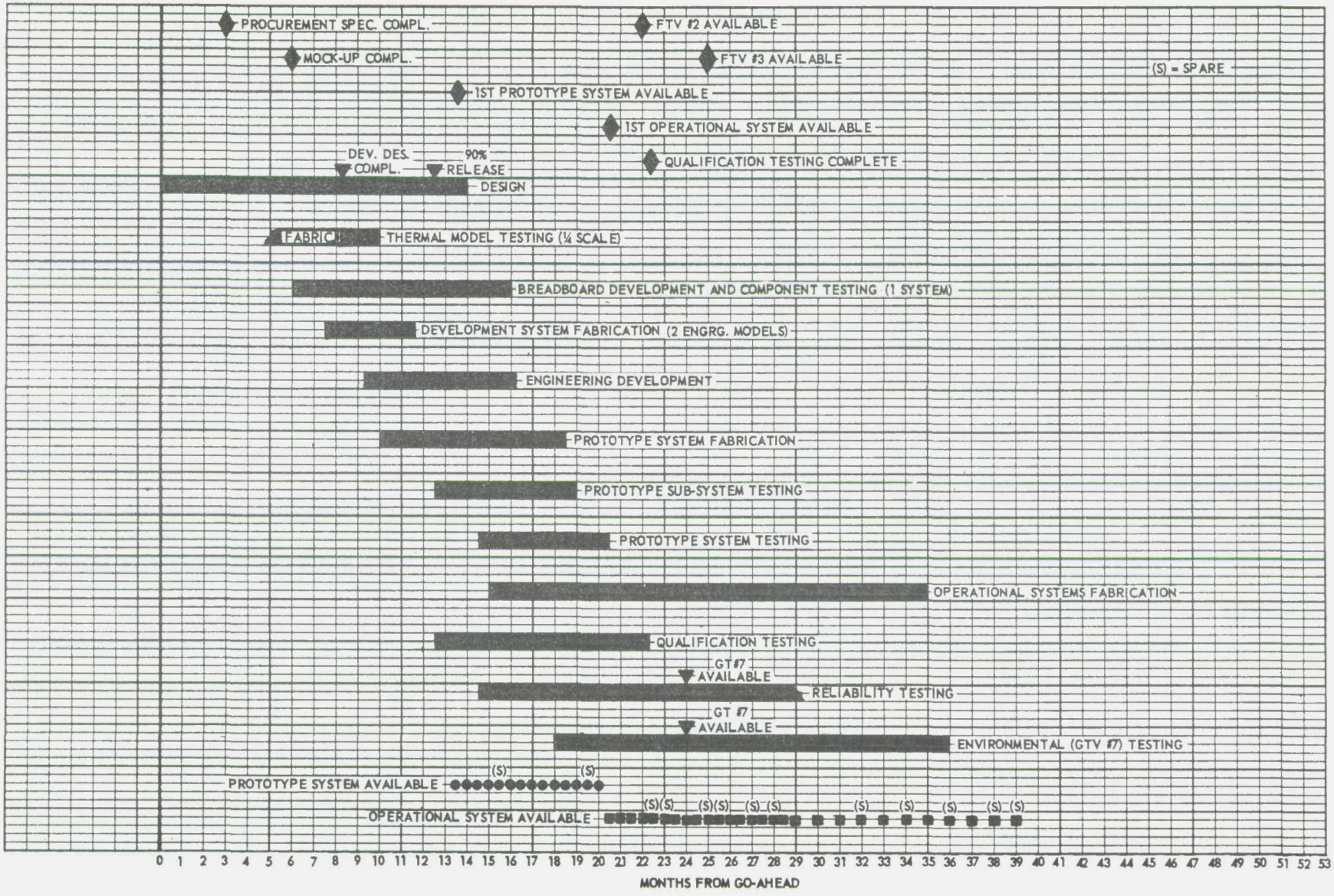


Figure 16–51 ENVIRONMENTAL CONTROL SYSTEM DEVELOPMENT SCHEDULE

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

CONCLUSIONS AND RECOMMENDATIONS

16.5 CONCLUSIONS AND RECOMMENDATIONS

The following conclusions and recommendations are made based on the results of the LEM environmental control system study described herein.

(a) The proposed ECS system will meet the requirements of the full range of LEM missions and can be developed and qualified within the time period required for the LEM development. The ECS is not considered a critical development item relative to the complete LEM program.

(b) A thorough analysis of all thermal control techniques for equipment and materials located outside the pressure shell was not made during this study program. Sufficient analysis was conducted to indicate that for the majority of this equipment, passive techniques (i. e., insulation and selective coatings) will provide suitable thermal control. A more detailed analysis is required to verify this initial work and to determine detailed requirements for individual items.

(c) A significant reduction in over-all vehicle weight can be obtained by staging waste material (i. e., urine, feces, etc.) on the lunar surface. It is recommended that more study be given this area to determine means of accomplishing this staging without contamination of the lunar surface.

(d) The weights for environmental equipment presented herein are considered to be conservative.

(e) It is recommended that more study be devoted to methods of combining functions of the LEM and Apollo ECS systems during the translunar phase.

BIBLIOGRAPHY

1. CVA Report E9R-11833 dated 15 January 1959, entitled 'Properties of Evaporative Fluids' by D. T. Cox.

2. ASD TR 61-162, Part I dated December 1961 entitled 'Analytical Methods for Space Vehicle Atmosphere Control Processes' by A. Shaffer.

3. AiResearch Report No. AE-1994R, Rev. 1 dated 3 April 1962, entitled 'Subcritical Cryogenic Storage System for Zero Gravity Operation' by J. S. Tyler.

4. Final Technical Report, Project No. 945-1, Contract NASw-150, Southwest Research Institute, entitled 'A Study of Trace Contaminants in Enclosed Systems' by H. C. McKee, et.al.

5. WADD TR 56-273 dated October 1956, entitled 'Handbook of Biological Data' by USAF Wright Air Development Center.

6. WADC TR 56-473 dated February 1957, entitled 'Final Report on Heat Storage Cooling of Electronic Equipment' by Wright Air Development Center.

7. WADC TR 59-253 dated November 1959, entitled 'Study of Equipment Cooling Systems' by Wright Air Development Center.

8. 'Liquid Transfer Media for Airborne Thermal Systems' by Tang, Chang, and Mason.

9. Project Apollo Working Paper No. 1015 dated 15 February 1961 by NASA Space Task Group, Langley Field, Virginia.

10. AEDC TR 59-11 dated June 1959, entitled 'Spreading of Rocket Exhausts at High Altitudes' by USAF Arnold Engineering Development Center.

11. Aerojet Report 2225B dated February 1962, entitled 'Space Vehicle Propulsion and Attitude Control' by Aerojet-General Corporation.

12. NASA TND-1095, entitled 'Interactions of Highly Underexpanded Jets with Simulated Lunar Surfaces' by Lewis Research Center, Cleveland, Ohio.

13. NASA TND-1017 dated February 1962, entitled 'An Exploratory Investigation of Jet-Blast Effects on a Dust Covered Surface at Low Ambient Pressure' by NASA Langley Research Center, Langley Air Force Base, Virginia.

14. Vought Astronautics Division Report AST/EOR-13156 dated September 1960, entitled 'Thermal And Atmosphere Control System Studies for Manned Spacecraft' by G. B. Whisenhunt, et.al.

15. WADD TN 60-161 dated July 1960, entitled 'Vapor Cycle Cooling Systems for Space Vehicles' by Wright Air Development Center.

APPENDIX

APPENDIX 16A

ANALOG ANALYSIS OF ATMOSPHERE CONTROL SYSTEMS

An analog computer was employed to assist in the analysis of the atmopshere control system. The analog was used to permit analysis of the many atmosphere control variables on a complete system basis. This Appendix presents a summary of this analog study.

The basic variable of any atmosphere control system for manned vehicles is man himself. In order to program man into the system, man's metabolic rates must be expressed mathematically. This was accomplished by using activity index (Z). Figure 16A-1 shows the definition of activity index. Man's metabolic rates (e.g., heat produced, oxygen consumed, carbon dioxide and water vapor given off), corresponding to each level of activity shown in Figure 16A-1 were obtained from Reference 5 and were programmed on the analog. To provide representative data for system analysis, two 24-hour activity cycles were defined for use during the analog runs. The first 24-hour cycle is shown in Figure 16A-2. This cycle was defined to include activity levels likely to be encountered both during lunar launch and during a typical period in lunar orbit. Figure 16A-3 shows the second 24-hour activity cycle which includes a normal period in orbit with an emergency situation suddenly imposed. When the emergency is over, the crew returns to the normal work-rest cycle.

A representative environmental control system was defined for analog analysis so that the data obtained would be as general as possible. A block schematic of this system is shown in Figure 16A-4. The equations, input parameters, constants, and output parameters which were used in the analog program are shown in the Attachment to the Appendix.

The components of the representative system include a three man cabin, an activated charcoal odor remover, a blower fan, a lithium hydroxide CO_2 absorber, a de-humidifier, a heat exchanger, and a supply of oxygen and nitrogen make-up gas.

The cabin atmosphere for the system studied consisted of oxygen and nitrogen at a nominal pressure of 7.5 psia. (This study was conducted prior to the decision on a pure oxygen system for the LEM). The nitrogen pressure was allowed to decrease gradually by leakage during flight, with nitrogen make-up gas being programmed for periodic manual addition. Total cabin pressure was controlled to $7.5 \pm .1$ psia by oxygen make-up gas. The oxygen flow rate using on-off control was 0.02 lb/min. during the periods of flow.

The lithium hydroxide (LiOH) CO_2 absorber was programmed to remove essentially all the CO_2 from the entering air. The LiOH canister would be changed manually by the crew at fixed time intervals.

Humidity and temperature control are accomplished by two heat exchangers and a water separator. The first heat exchanger lowers the air temperature to 35 - 45°F condensing water out of the airstream. The free water in the air is then removed by the water separator. Air leaves the

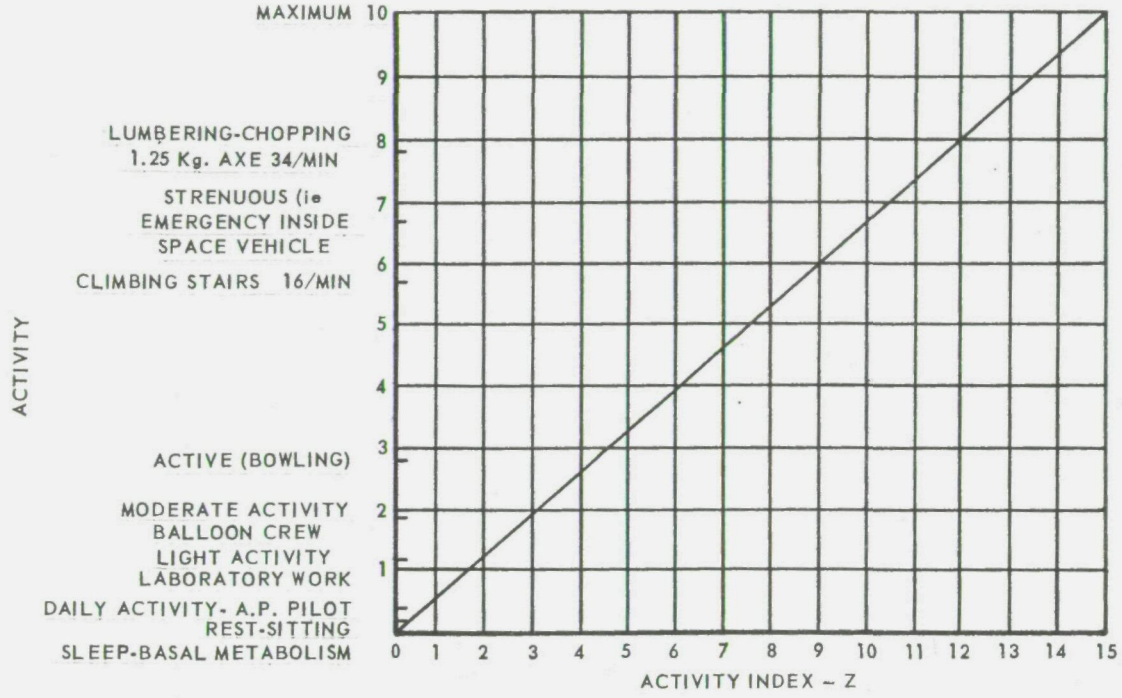


Figure 16A-1 DEFINITION OF ACTIVITY INDEX

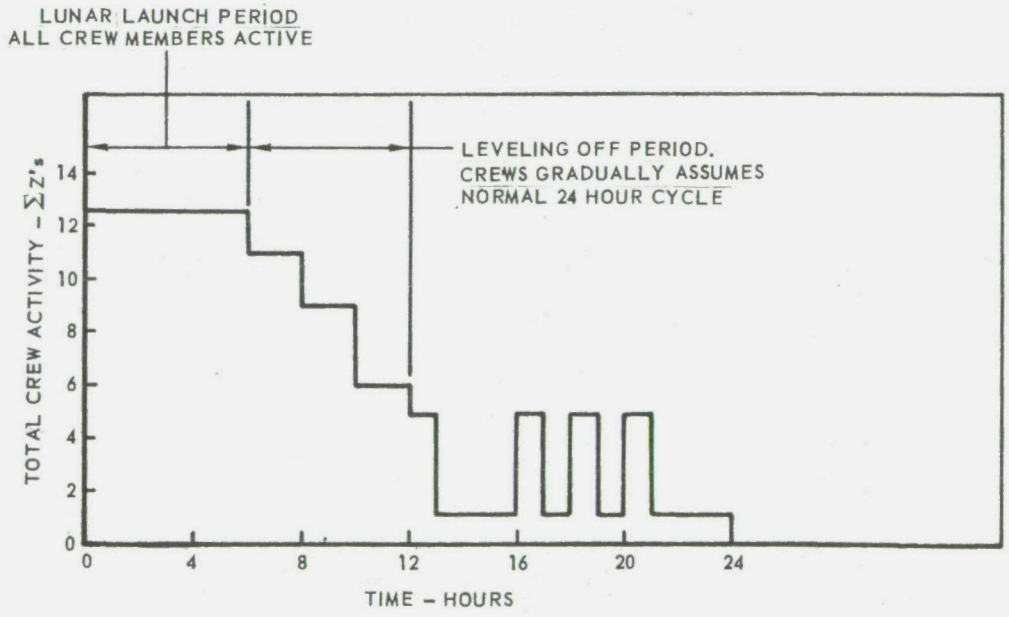


Figure 16A-2 PROGRAMMED ACTIVITY CYCLE FOR TWENTY-FOUR HOUR RUNS

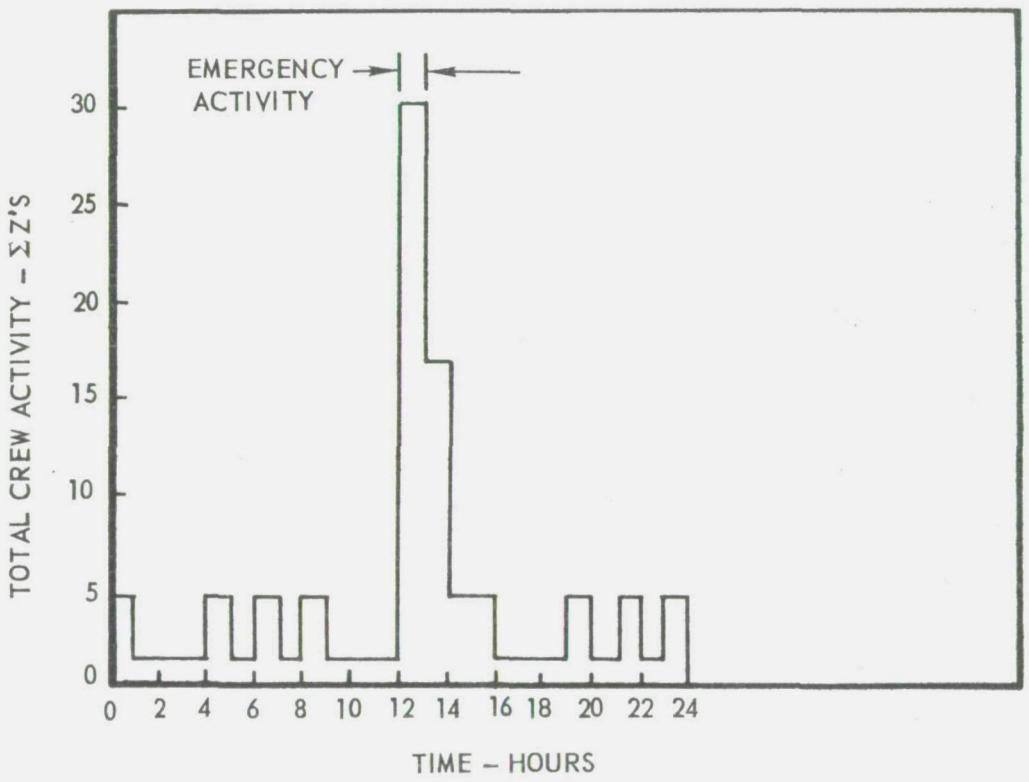


Figure 16A-3 PROGRAMMED ACTIVITY CYCLE FOR "EMERGENCY" ACTIVITY RUNS

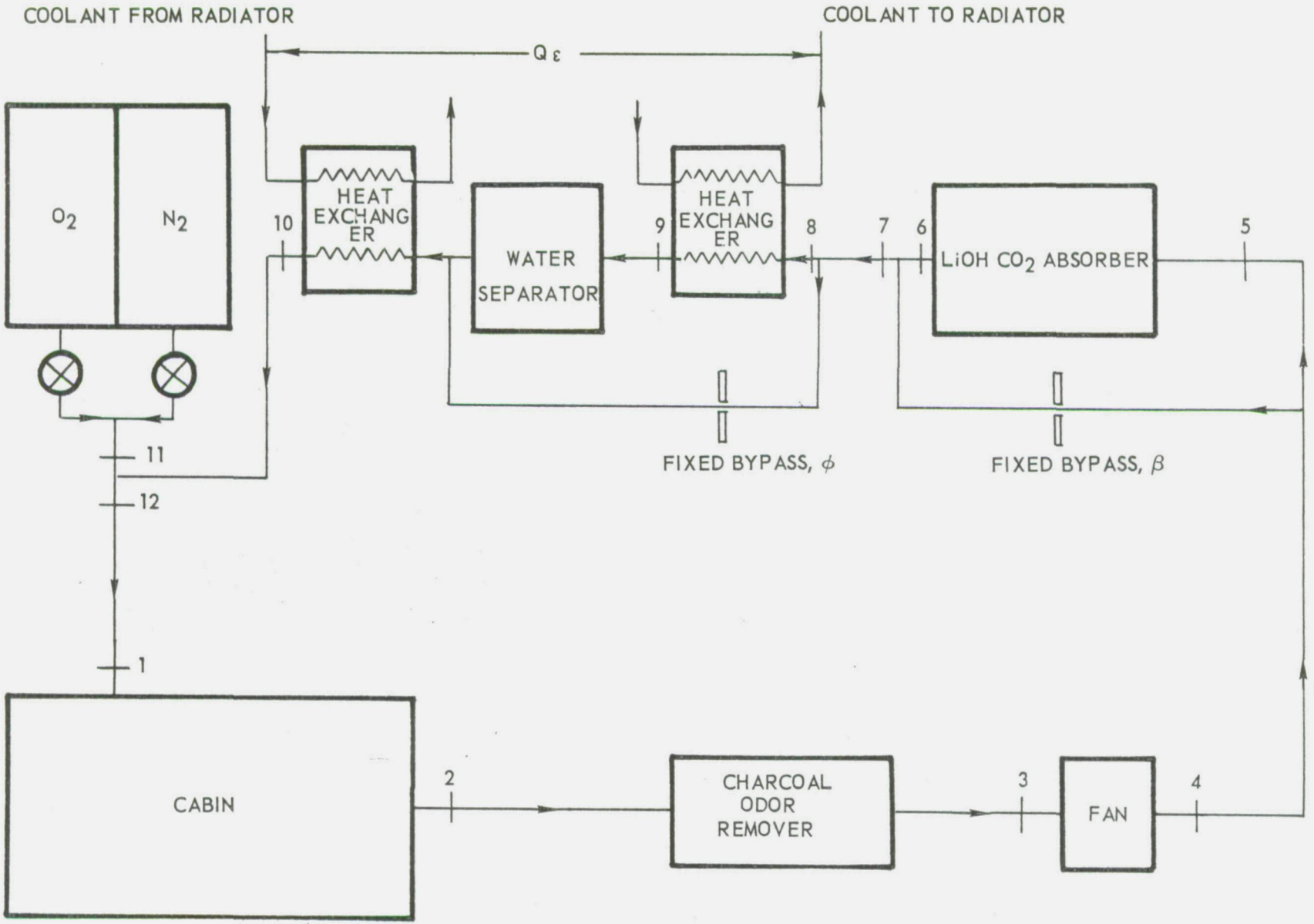


Figure 16A-4 SCHEMATIC OF REPRESENTATIVE ATMOSPHERE CONTROL SYSTEM

ASTRONAUTICS DIVISION
CHANCE VOUGHT CORP

water separator at 35 - 45°F and 100% relative humidity and is mixed with the air by-passed around the separator. After mixing, the air flows through the second heat exchanger and into the cabin. The cabin air temperature is maintained at a constant value of 75°F by the addition or removal of heat in the second heat exchanger. The analog output parameter, Q_E, is the algebraic sum of the heat transferred in the two exchangers.

A typical analog trace of one day's operation of the atmopshere control system studied is shown in Figure 16A-5. The activity cycle shown in Figure 16A-2 was used during this test. In Figure 16A-5 the CO_2 partial pressure is shown to approach a level of 0.07 psia (approximately 1% by volume) during the period of high activity. During the work-rest cycle, the CO_2 partial pressure in the cabin varies between 0.02 and 0.04 psia. The maximum water vapor partial pressure is 0.27 psia during launch, and the variation is between 0.12 and 0.18 psia during the normal work cycle. This corresponds to a relative humidity range of 28 to 63 per cent. The nitrogen partial pressure remained essentially constant throughout the run because of the relatively low leakage rate used. Since a large portion of the heat removed comes from the sensible heat liberated by the crew, the heat removed followed the activity cycle inputs very closely. The latent heat of the water removed caused a slight lag in the heat removed trace. The power required to drive the fan remained essentially constant throughout the run. The carbon dioxide removal and water generation rates in the absorber are seen to be proportional to the CO_2 concentration. Water removal in the de-humidifier is proportional to the difference between the water vapor partial pressure entering the heat exchanger and the saturation pressure of the water leaving. Therefore, the water removal rate is a function of the water vapor partial pressure and saturation temperature in the condenser. A variation in the oxygen partial pressure between approximately 3.1 and 3.3 psia occured because of the on-off characteristics of the oxygen make-up system. The total pressure was controlled to $7.5 \pm .1$ psia by the oxygen make-up. The small variation is hardly noticeable on the total pressure trace. The occasional peaks in the CO_2 removal and water removal traces are due to noise in the computer system. The oxygen make-up flow alternated from 0.02 lb/min., to zero, and back to 0.02 lb/min. as required to keep the total pressure within limits. At the time of launch (zero time on the trace), the total pressure is seen to be slightly below the desired value. Oxygen make-up gas flowed into the system until the total pressure was built up to 7.6 psia. Then the flow ceased until the total pressure dropped to 7.4 psia. At this pressure, the oxygen flow resumed and the process repeated itself. The percentage of time the oxygen is flowing is much greater during launch and the period shortly afterwards because the higher activity level of the crew results in a higher oxygen consumption rate.

Should an accident occur causing an emergency, the activity level of the crew members would increase sharply. The increased rate at which heat is given off, oxygen consumed, and water and CO_2 released, would throw a large load on the atmosphere control system. The effects of a sudden emergency are shown in Figure 16A-6. Both the H_2O and CO_2 partial pressures increase significantly. The relative humidity reached approximately 100% during this test. The oxygen partial pressure dropped slightly during the emergency because the increased H_2O and CO_2 partial pressures kept the total pressure high for a period of time thereby cutting off oxygen make-up

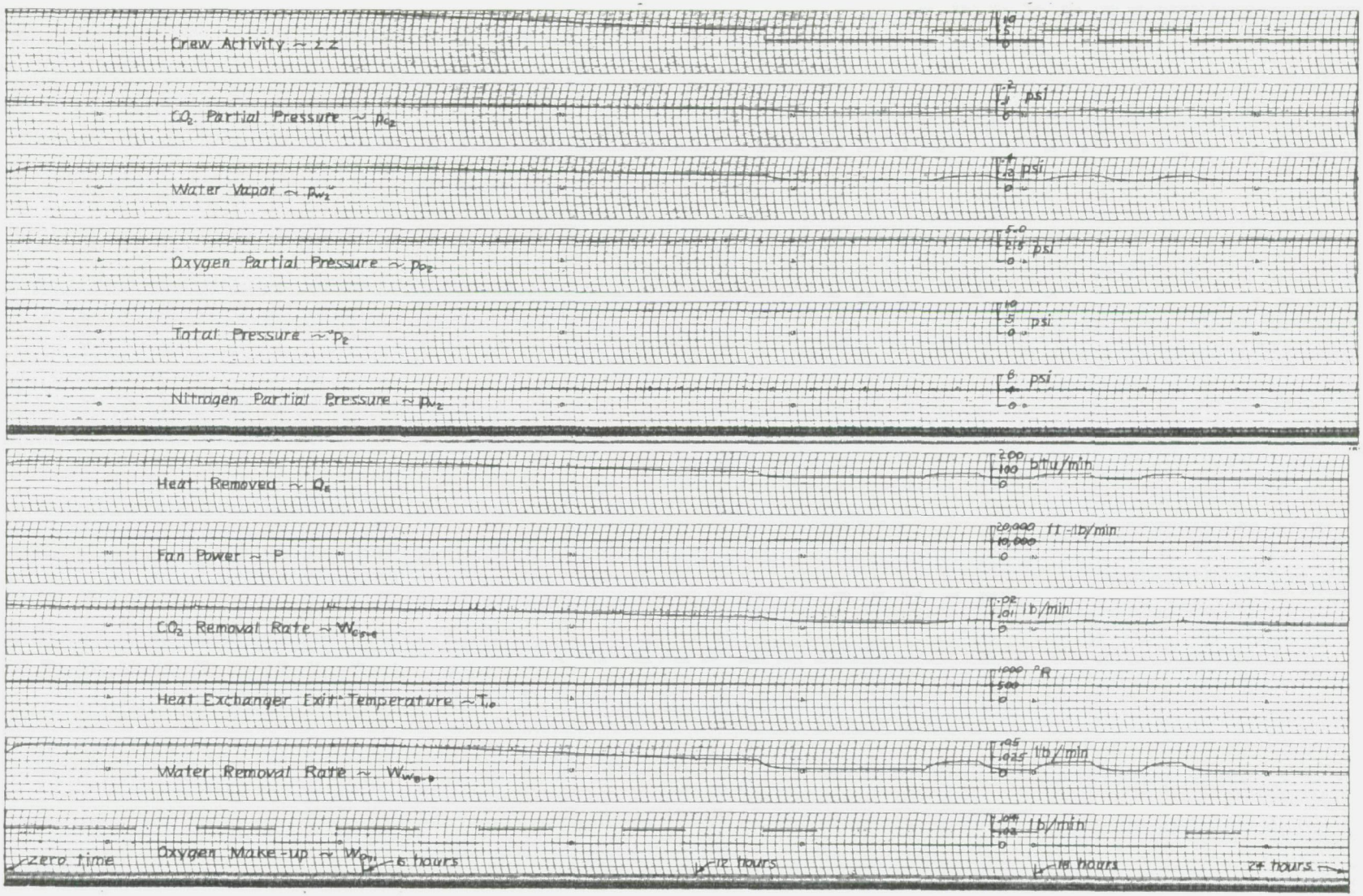


Figure 16A-5 ANALOG PERFORMANCE OF REPRESENTATIVE SYSTEM

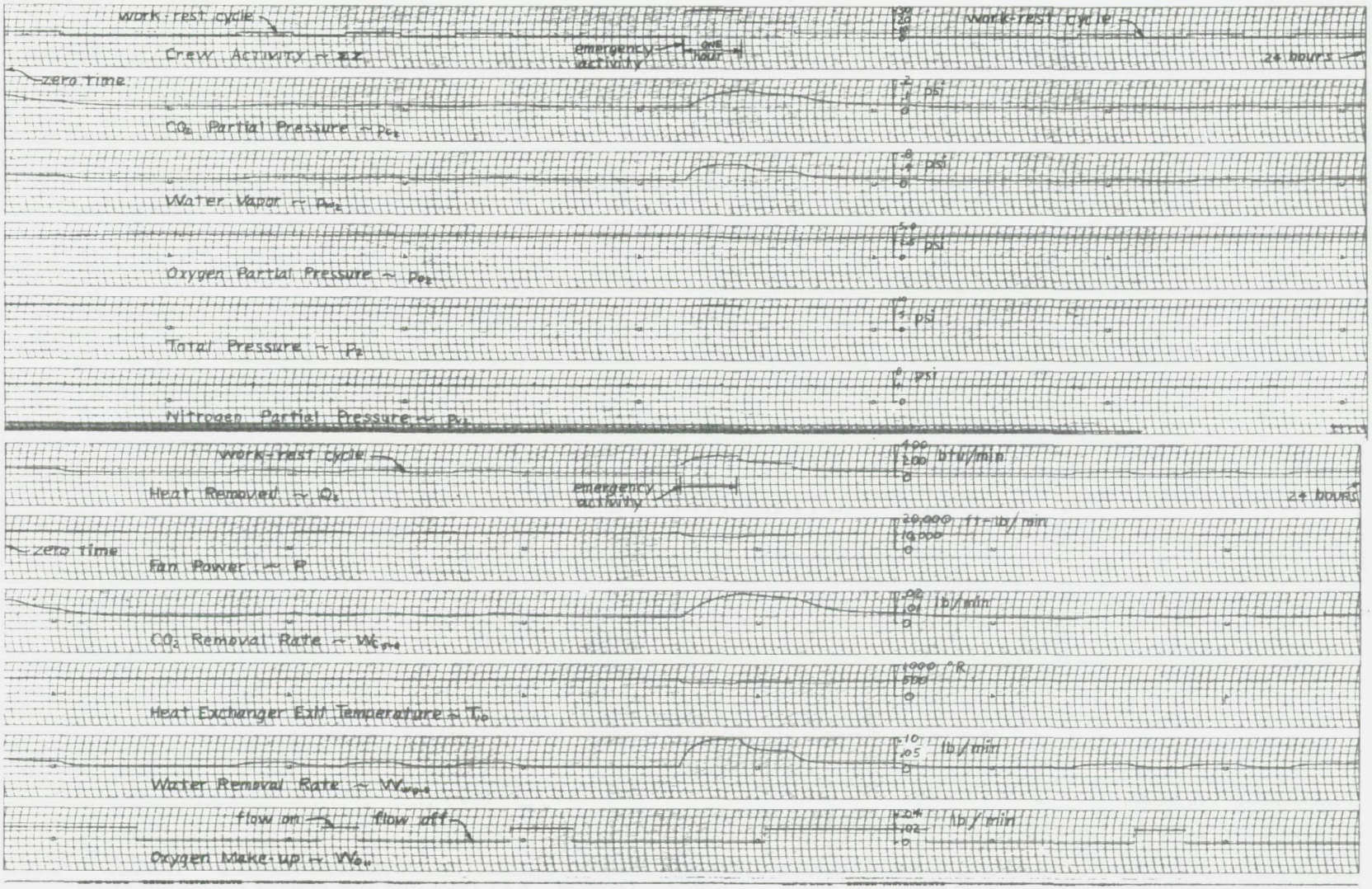


Figure 16A-6 ANALOG PERFORMANCE OF REPRESENTATIVE SYSTEM (EMERGENCY ACTIVITY CYCLE)

flow. The heat removed from the heat exchangers increased from 40 to 260 BTU/minute. The fan power dropped slightly due to the flow of cooler air from the exchanger to the cabin. The required temperature of the air leaving the heat exchanger and entering the cabin dropped to about 400°R. This is an unreasonably low number and indicates the flow rate used was too low. During the emergency this system allowed atmosphere control parameters to reach upper limits and in some cases to exceed upper limits. This is undesirable since the emergency already taxes the crew capabilities. The design parameters were varied in order to arrive at a workable system.

Some of the characteristics of the system studied are shown in Figures 16A-7 through 16A-9 as a function of flow rate, cabin total pressure, and wall temperature. The magnitude of the parameters will change with system sizing, but their characteristics will be the same. Figure 16A-7b shows the power, P, required to drive the blower fan versus flow ra e through the system. The variation in CO_2 pressure, pc_2, with flow rate is shown in Figure 16A-7a. The upper curve, $\sum Z_i = 12$, corresponds to the maximum activity expected during a mission except for an emergency. The lower curve, $\sum Z_i = 1$, corresponds to the minimum activity. Figure 16A-7d shows the water vapor partial pressure, p_{w2}, as a function of flow rate. The pressures corresponding to a relative humidity of 25% and 55% are also shown. In Figure 16A-7c the heat removed, from the heat exchangers, Q_E, is shown versus flow rate for maximum and minimum activity. Figure 16A-8 shows the CO_2 concentration, fan power, heat removed, and water vapor partial pressure as a function of cabin total pressure for maximum and minimum activity. The heat removed, Q_E, and the temperature of the air leaving the heat exchanger, T_{10}, are very sensitive to varying wall temperatures. The wall temperature described here is that of the inside wall of the cabin. The effect of wall temperature on heat removed and heat exchanger temperature are shown in Figure 16A-9.

These analog studies have produced considerable information on the operation of the complete atmosphere control system. Some of the conclusions resulting from these studies are:

(a) For a constant air flow system, carbon dioxide concentration can be controlled within allowable limits without complex controls.

(b) For a constant flow system, relative humidity may be controlled within desirable limits by controlling the liquid inlet temperature from the radiator to a constant level.

(c) A significant variation in the rate at which heat must be rejected from the cabin heat exchanger occurs as crew activity changes.

(d) The heat load establishes the minimum air flow through the atmosphere control system. A separate heat exchanger for cabin temperature control is desirable.

(e) The heat load is very sensitive to changes in cabin wall temperatures. This means that adequate insulation of the cabin wall is required.

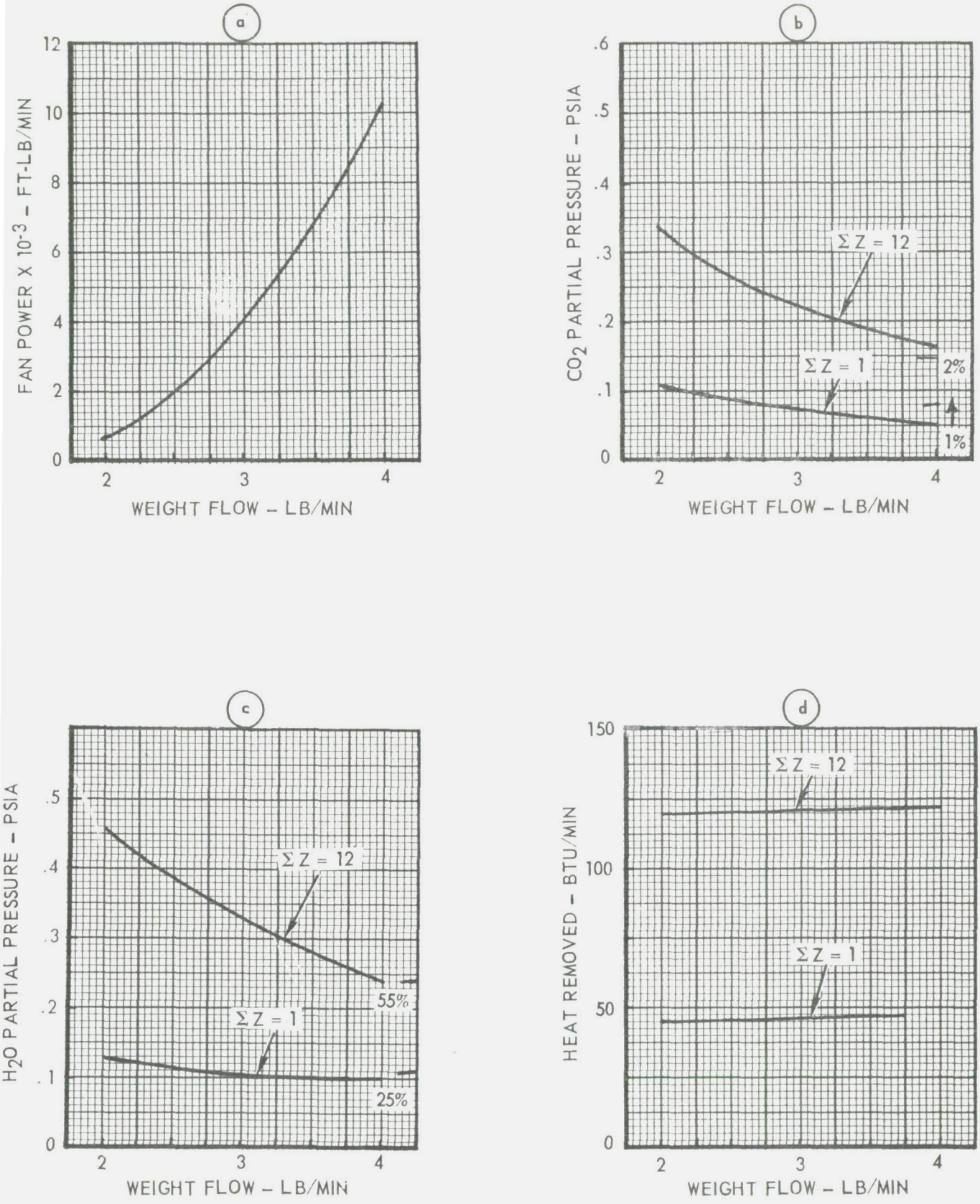


Figure 16A-7 EFFECT OF VARYING WEIGHT FLOW ON SYSTEM PERFORMANCE

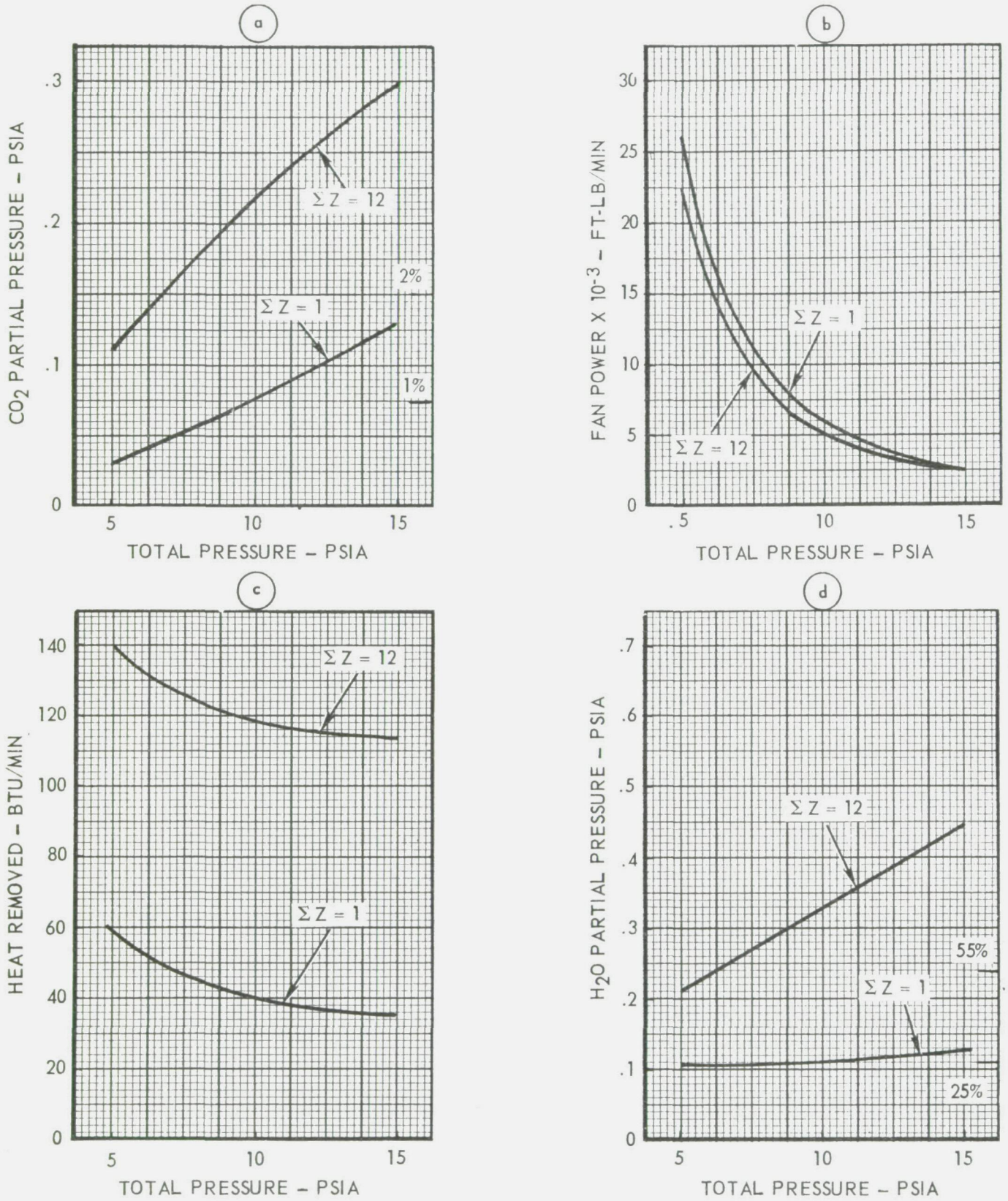


Figure 16A-8 EFFECT OF VARYING TOTAL PRESSURE ON SYSTEM PERFORMANCE

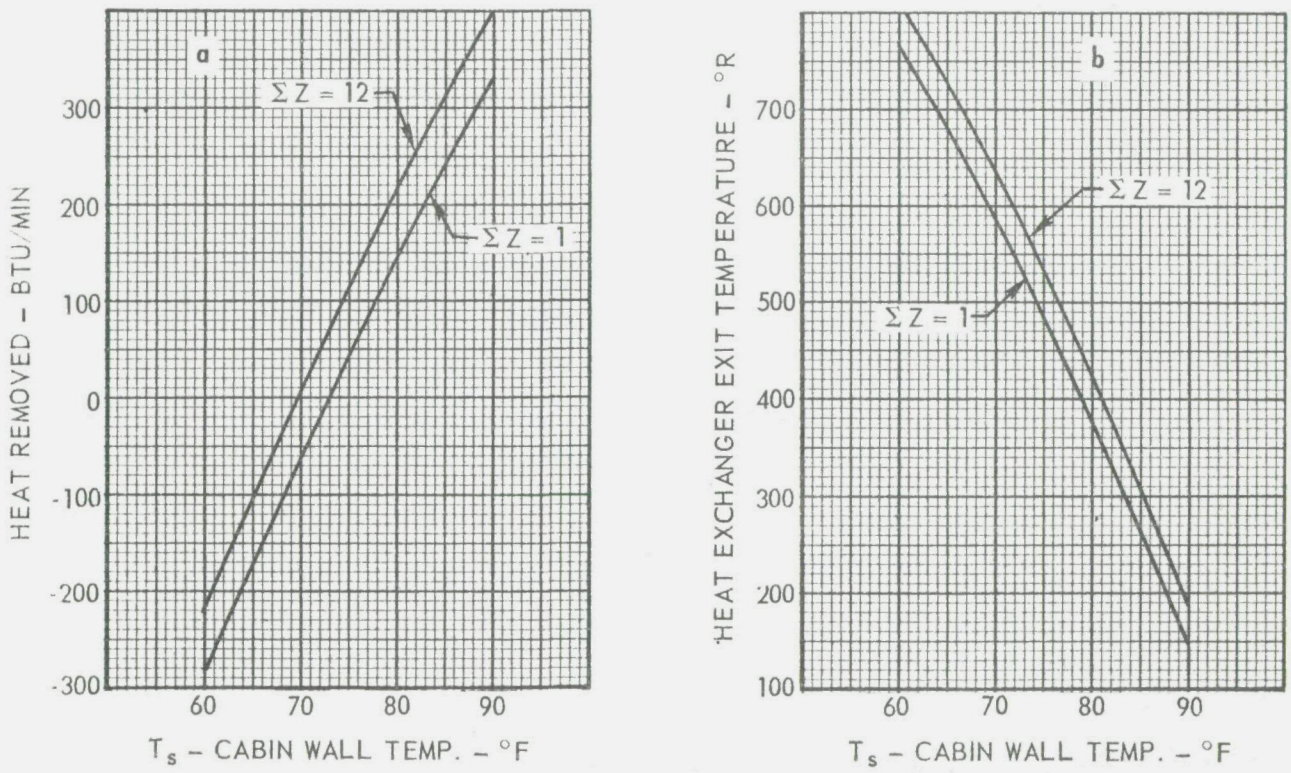


Figure 16A-9 EFFECT OF WALL TEMPERATURE ON SYSTEM PERFORMANCE

ATTACHMENT TO APPENDIX

DEVELOPMENT OF EQUATIONS FOR ATMOSPHERE CONTROL SYSTEM ANALOG

DEFINITION OF SYMBOLS

Basic Symbols

A, B, C	Constants in the equations for heat liberated by the crew
E, F, U	Constants in the equations for water vapor released by the crew
H	Fan pressure ratio
h	Coefficient for Q_s
J	Weight fraction of CO_2 absorbed in absorber
K	Pressure drop coefficient
k	Ratio of specific heats
L	Latent heat of vaporization (for O_2 and N_2, L includes $C_p \Delta T$)
P	Theoretical fan power
p	Pressure
p_O, p_W, etc.	Partial pressures of atmospheric gases
Q	Heat absorbed or removed
R	Gas constant
T	Temperature
V	Cabin volume
W	Flow rate
W_C, W_O, etc.	Rate of removal or addition of constituent gases
X	Weight fraction of constituent gas
Y	Cabin circulation coefficient
Z	Activity index
C_p	Specific heat
θ	Time

ANALOG PROGRAM FOR ATMOPSHERE CONTROL SYSTEM

Analog Equations - The following equations describe the operation of the atmosphere control system analyzed. Figure 16A-10 presents a circuit diagram of the analog network.

1. Cabin

a. Crew

(1) Heat Given Off - For the temperature range under consideration, heat is given off proportional to the person's activity (Z) and the ambient temperature (T_2). Developing these relationships gives an equation of the form:

$$Q_H = (1 \times AZ)(B+CT_2) \text{ BTU/man-min. for one man}$$

$$Q_H = (3+A[Z_i+Z_j+Z_k])(B+CT_2) \text{ BTU/min. for a three man crew}$$

(2) Oxygen Consumed - For the average diet, each pound of oxygen consumed liberates approximately 6600 BTU.

$$W_{O_{1-2}} = \frac{QH}{6600} \text{ lbs/min.}$$

(3) Carbon Dioxide Given Off - Carbon dioxide given off is proportional to the oxygen consumed. For a respiratory quotient of .85

$$W_{C_{1-2}} = 1.17\, W_{O_{1-2}} \text{ lbs/min.}$$

(4) Water Vapor Released - As in the case of the heat given off, the water vapor released is also proportional to the person's activity and the ambient temperature.

$$W_{W_{1-2}} = (1+EZ)(F+UT_2) \text{ lbs/man min. for one man}$$

$$W_{W_{1-2}} = (3+E[Z_i+Z_j+Z_k])(F+UT_2) \text{ lbs/min. for a three man crew}$$

b. Heat Transfer to Cabin Wall - Assuming the temperature difference is small and air circulation is proportional to the flow entering the cabin (W_1), the heat transfer is proportional to the flow rate and difference between the wall temperature (T_S) and the ambient temperature (T_2).

$$Q_S = h\,(1+yW_1)(T_2-T_S) \text{ BTU/min.}$$

c. Leakage - There is some leakage flow (W_L) from the cabin. Each component gas is lost according to its proportion by weight (X) of the cabin atmosphere.

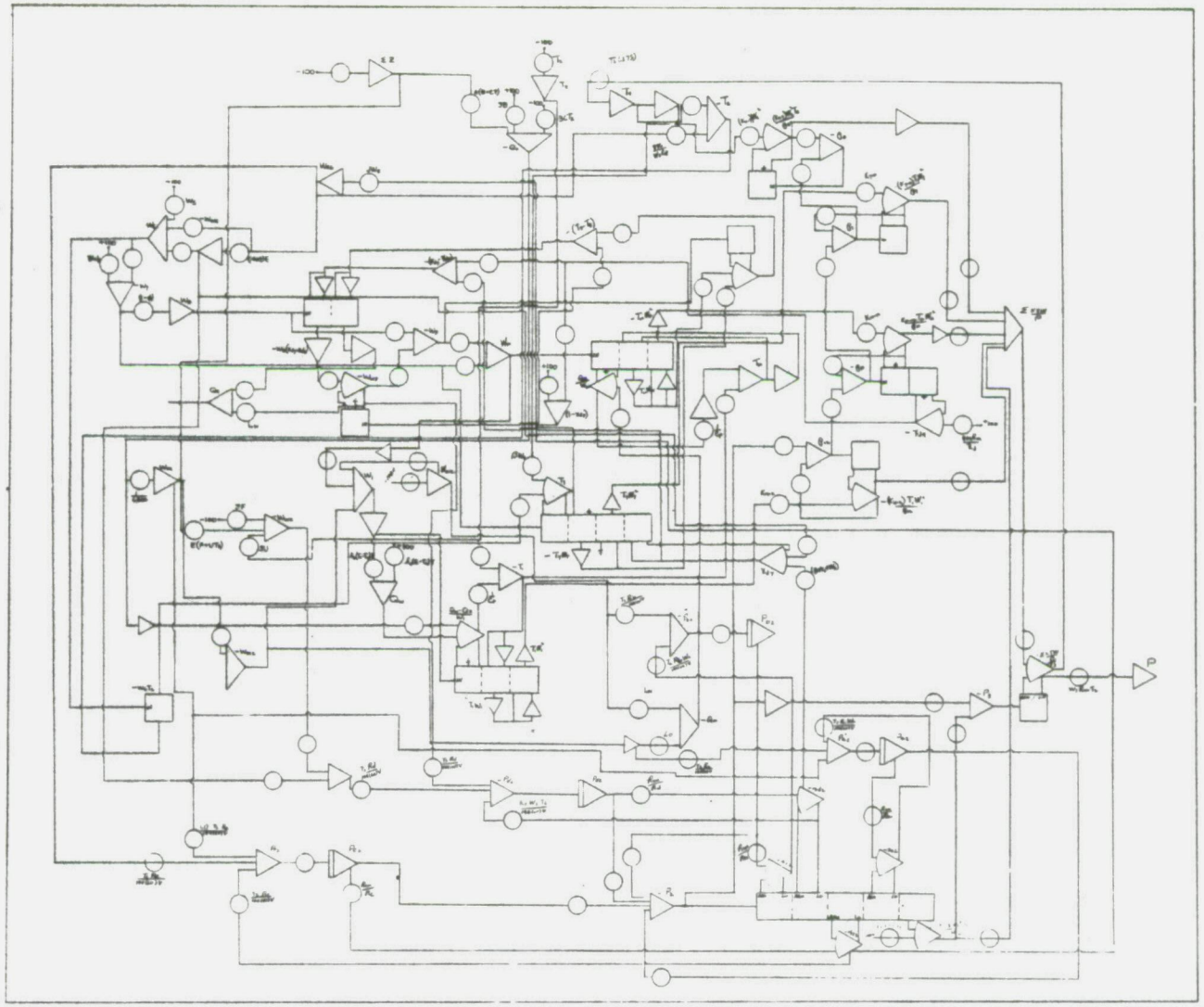


Figure 16–A10 DIAGRAM OF THE ANALOG PROGRAM FOR THE ATMOSPHERE CONTROL SYSTEM

CHANCE VOUGHT CORP
ASTRONAUTICS DIVISION

Oxygen	$X_{o_2} W_L$ lbs/min.
Nitrogen	$X_{n_2} W_L$ lbs/min.
Carbon Dioxide	$X_{c_2} W_L$ lbs/min.
Water Vapor	$X_{w_2} W_L$ lbs/min.

d. Temperature - The cabin temperature is maintained at a predetermined value by adjusting the temperature of the gas leaving the heat exchanger.

e. Partial Pressures - The partial pressures are found from a flow balance of the constituents in the system.

$$p_{o_2} = p_o \text{ initial} + \frac{T_2 R_o}{V} \int W_{o_{5-6}} + W_{o_{11}} - W_{o_{1-2}} - X_{o_2} W_L$$

$$p_{n_2} = p_n \text{ initial} + \frac{T_2 R_n}{V} \int W_{n_{11}} - X_{n_2} W_L$$

$$p_{c_2} = p_c \text{ initial} + \frac{T_2 R_c}{V} \int W_{c_{1-2}} - W_{c_{5-6}} - X_{c_2} W_L$$

$$p_{w_2} = p_w \text{ initial} + \frac{T_2 R_w}{V} \int W_{w_{1-2}} - W_{w_{8-9}} - X_{w_2} W_L + W_{w_{5-6}}$$

The total pressure is equal to the sum of the partial pressures.

$$p_2 = p_{o_2} + p_{n_2} + p_{c_2} + p_{w_2}$$

f. Composition - The weight composition of the gases is found by converting partial pressure ratios into weight ratios.

$$X_{o_2} = p_{o_2} R_m / p_2 R_o \text{ etc.}$$

It is not necessary to calculated X_{n_2} since $\Sigma X's = 1$.

2. Fan

a. Pressure Ratio - The pressure ratio is calculated by:

$$\frac{p_4}{p_3} = 1 + \frac{\sum \frac{KWT}{p}}{p_3}$$

b. Power - The fan power is calculated by:

$$P = \frac{k\, W_2\, R_m\, T_2}{k-1}\left[\left(\frac{p_4}{p_3}\right)^{\frac{k-1}{k}} - 1\right]$$

for small ΔP this may be approximated by:

$$P = W_2\, R_m\, T_2\left(\frac{\sum \frac{KWT}{p}}{p_3}\right) \text{ ft. lbs/min.}$$

c. Temperature Rise - The temperature rise across the fan is calculated by:

$$T_4 = T_2\left[\left(\frac{p_4}{p_3}\right)^{\frac{k-1}{k}}\right]^{\circ}R$$

For p_4/p_3 near 1, the quantity $\left(\frac{p_4}{p_3}\right)^{\frac{k-1}{k}}$ may be approximated by linear function of p_4/p_3

$$T_4 = T_2\left[1 + .275\ \frac{p_4}{p_3}\right]$$

3. Carbon Dioxide Absorber

a. Carbon Dioxide Absorber - If the absorber is properly designed, it will absorb practically all the carbon dioxide entering ($X_{c_2}\ W_5$). Provisions are included for absorption of some fraction, J, of the CO_2 entering the absorber.

$$W_{c_{5-6}} = JX_{c_2}\ W_5 \text{ lbs/min.}$$

b. Heat Released - Lithium hydroxide liberates approximately 876 BTU per pound of carbon dioxide absorbed.

$$Q_A = 876\ W_{c_{5-6}} \text{ BTU/min.}$$

c. Water Vapor Given Off - For each pound of carbon dioxide absorbed, 0.409 pounds of water vapor are given off. A fraction $(1 - \epsilon)$ of the water vapor may be reabsorbed by the lithium hydroxide. The net amount given off is then

$$W_{w_{5-6}} = 0.409\ \epsilon\ W_{c_{5-6}} \text{ lbs/min.}$$

d. Composition - The composition of the gas leaving the absorber is calculated by a mass balance.

$$W_6 = W_5 - W_{c5-6} \pm W_{w5-6} + W_{o5-6}$$

$$X_{o6} = \frac{X_{o2} W_5 + W_{o6}}{W_6}$$

$$X_{c6} = \frac{(1-J)(X_{c_2} W_5)}{W_6}, \quad X_{w6} = \frac{X_{w_2} W_5 \quad W_{w5-6}}{W_6}$$

e. Temperature - The temperature of the gas is increased because of the heat released in the absorber.

$$T_6 = T_4 + \frac{Q_A}{W_5 C_p} \ ^oR$$

4. Mixing of By-Passed Gas After the Absorber - After leaving the absorber, the gas is mixed with that by-passed. Assuming the compositions are similar, simple mixing equations may be used.

$$T_1 = \frac{\beta W_2 T_4 \quad W_6 T_6}{W_7} \ ^oR$$

$$W_7 = \beta W_2 + W_6 \text{ lbs/min.}$$

$$X_{o7} = \frac{X_{o2} \beta \ W_2 \quad X_{o6} W_6}{W_7} \quad \text{etc.}$$

5. Humidifier and Heat Exchanger

a. Water Vapor - The partial pressure of the water vapor leaving the humidifier-heat exchanger is constant corresponding to some saturation temperature. The fraction (X_{w9}), of water vapor leaving can then be calculated:

$$X_{w9} = \frac{p_{w9} R_m}{p_9 R_w}$$

From a mass balance, the amount of water removed (W_{w8-9}) can be calculated.

$$W_{w8-9} = \frac{(X_{w7} - X_{w9})}{(1 - X_{w9})} W_8 \text{ lbs/min.}$$

b. Heat Removed and Temperature - The temperature leaving the humidifier-heat exchanger is varied to maintain the cabin temperature constant. The net heat removed (Q_E) is that of condensing the water vapor removed and changing the temperature of the gas.

$$Q_E = W_7 C_p (T_8 - T_{10}) + W_{w8-9} L_w \text{ BTU/min.}$$

also

$$Q_E = W_8 C_p (T_7 - T_9) + W_{w8-9} L_w \text{ BTU/min.}$$

Note: A part of the flow, $\emptyset$, may be by-passed around the water condensing unit, but all the flow goes through the heat exchanger to control temperature of the gas entering the cabin.

6. Mixing of By-Passed Gas After Humidifier-Heat Exchanger

a. Composition - The composition after mixing is calculated by a mass balance between the exchanger and by-pass flows.

$$W_{10} = W_7 - W_{w8-9}$$

$$X_{o10} = X_{o7} W_7/W_{10}$$

$$X_{c10} = X_{c7} W_7/W_{10}$$

$$X_{w10} = \frac{X_{w7} W_7 - W_{w8-9}}{W_{10}}$$

b. Temperature - The temperature of the gas leaving the exchanger is calculated from a simple mixing relation.

$$T_9 = \frac{T_{10} W_{10} - \emptyset W_7 T_7}{W_9} \ ^oR$$

7. Gas Make-Up

a. Oxygen - Oxygen is supplied to keep the oxygen partial pressure constant. The oxygen supplied must equal the total of oxygen consumed and lost through leakage.

$$W_{o11} = W_{o1-2} + X_{o2} W_L - W_{o5-6}$$

b. Nitrogen - Nitrogen is supplied to keep the total pressure constant. Since the nitrogen partial pressure may be calculated from the total pressure, oxygen, carbon dioxide, and water vapor partial pressures, the nitrogen make-up is not calculated.

c. Heat Removed - It is assumed the oxygen and nitrogen make-up are stored as liquids. The heat (Q_m) is required to vaporize the fluid.

$$Q_m = L_n W_{n_{11}} + L_o W_{o_{11}} \text{ BTU/min.}$$

d. Temperature - The gas flowing through the gas make-up is cooled by Q_m. Calculating the required temperature before entry to the gas make-up,

$$T_{10} = T_1 + \frac{Q_m}{W_{10} C_p} \ ^oR$$

e. Composition - The composition of the gas leaving the make-up is calculated by a mass balance.

$$W_1 = W_{10} + W_{o11} + W_{n11} \text{ lbs/min.}$$

$$X_{o_1} = \frac{X_{o_{10}} W_{10} + W_{o11}}{W_1}$$

$$X_{c_1} = X_{c_{10}} W_{10}/W_1$$

$$X_{w_1} = X_{w_{10}} W_{10}/W_1$$

8. Gas Temperature Entering Cabin

The temperature of the gas entering the cabin required to keep the cabin temperature constant is determined by the net heat transferred in the cabin.

$$T_1 = T_2 + \frac{Q_S - Q_H}{W_1 C_p} \ ^oR$$

9. Oxygen Make-Up

The oxygen make-up flow controls the total pressure to 7.5 ± .1 psia. If the total pressure is less than 7.4, the oxygen make-up is .02 lb/min. The flow continues until the total pressure is built up to

7.6 psia and then stops. No flow occurs until the pressure drops to 7.4 psia. In equation form, this may be written:

For $p_2 \leq 7.4$, $W_{o_{11}} = .02$

$W_{o_{11}} = .02$ until $p_2 = 7.6$

For $p_2 \geq 7.6$, $W_{o_{11}} = 0$

$W_{o_{11}} = 0$ until $p_2 \leq 7.4$

Table 16-AI lists the required input data, as well as the output information, for use of the analog program.

TABLE 16A-I ANALOG PROGRAM INPUTS AND OUTPUTS

INPUTS AND OUTPUTS

INPUTS	OUTPUTS
P_{n_o}	P_{c_2}
T_2	P_{w_2}
P_o	P_{n_2}
W_L	$W_{o_{11}}$
W_2	$W_{w_{5-6}}$
ϕ	$W_{w_{8-9}}$
β	$W_{c_{5-6}}$
Σ Z's	P
V	Q_E
T_s	T_{10}
P_{c_o}	P_2
P_{w_o}	Σ Z's
P_{w9}	

CONSTANTS

SYMBOL	VALUE	SYMBOL	VALUE
A	82	R_w	86 ft/^oR
B	23 BTU/min.	R_n	55 ft/^oR
C	-.034 BTU/min-^oR	J	.98
E	7.12	ϵ	.95
F	-.00708 lbs/min.	L_n	90 BTU/lb
U	.000014 lb/min-^oR	L_o	100 BTU/lb
h	10 BTU/min-^oR	L_w	1070 BTU/lb
k	1.4	K_{2-3}	.000044 $\frac{min^2}{^oR\ ft^4}$
y	.25 min/lb	K_{12-2}	.00035 "
C_p	.24 BTU /lb ^oR	K_{10-12}	.0000087 "
R_m	52 ft/^oR	K_{7-10}	.000175 "
R_o	48 ft/^oR	K_{4-7}	.000087 "
R_c	35 ft/^oR		

ABSTRACT

14172

The LEM Environmental Control System study results and recommendations are presented. Subsystem trade studies, system parametric studies, analysis of special problems and a description of the operation of the recommended system are included. The recommended system provides thermal and atmosphere control of the LEM for a 24 hour lunar cold side mission with provisions for growth to a 7-day cold side mission or a 2-day hot side mission. Where possible, proven components have been incorporated. It is concluded that the recommended system meets the LEM requirements and that the system can be developed in the required time schedule.

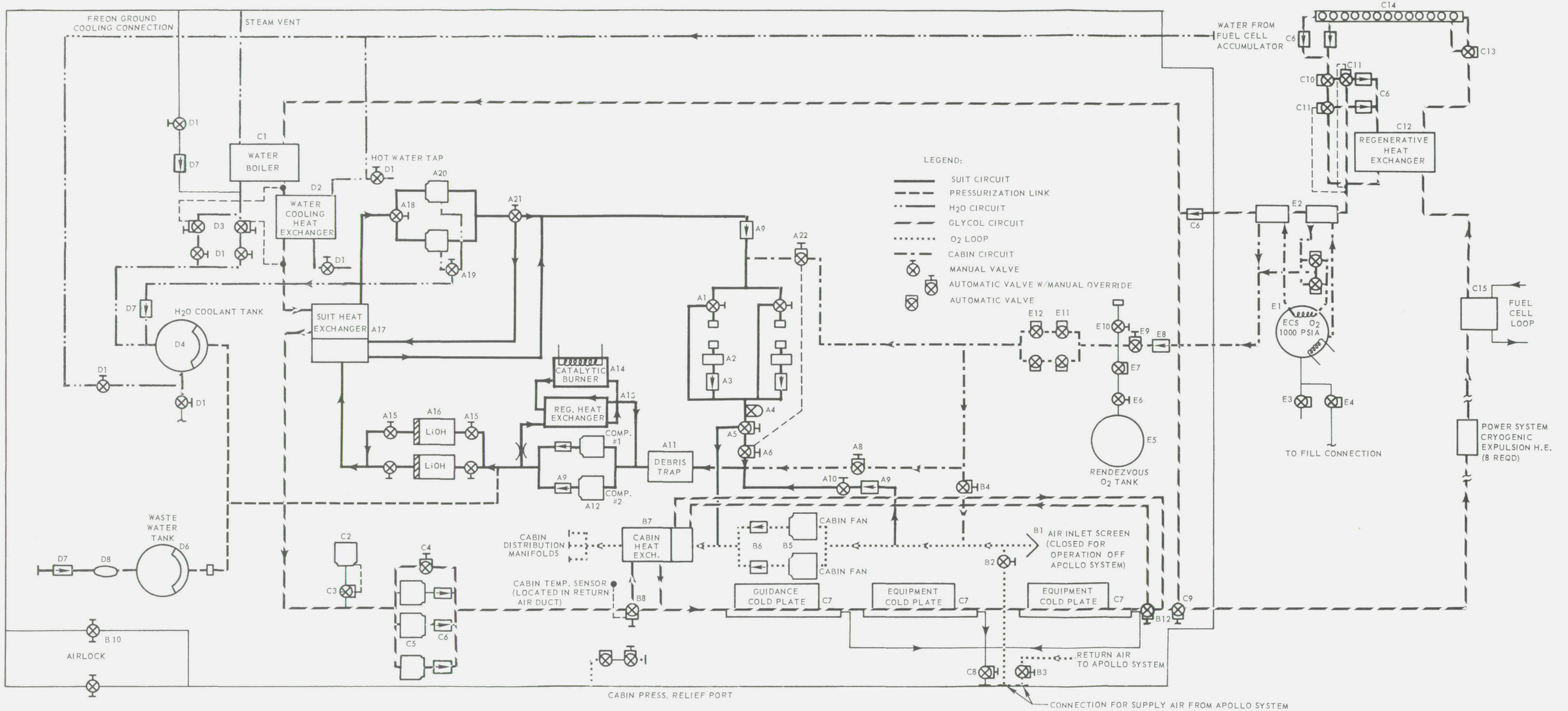
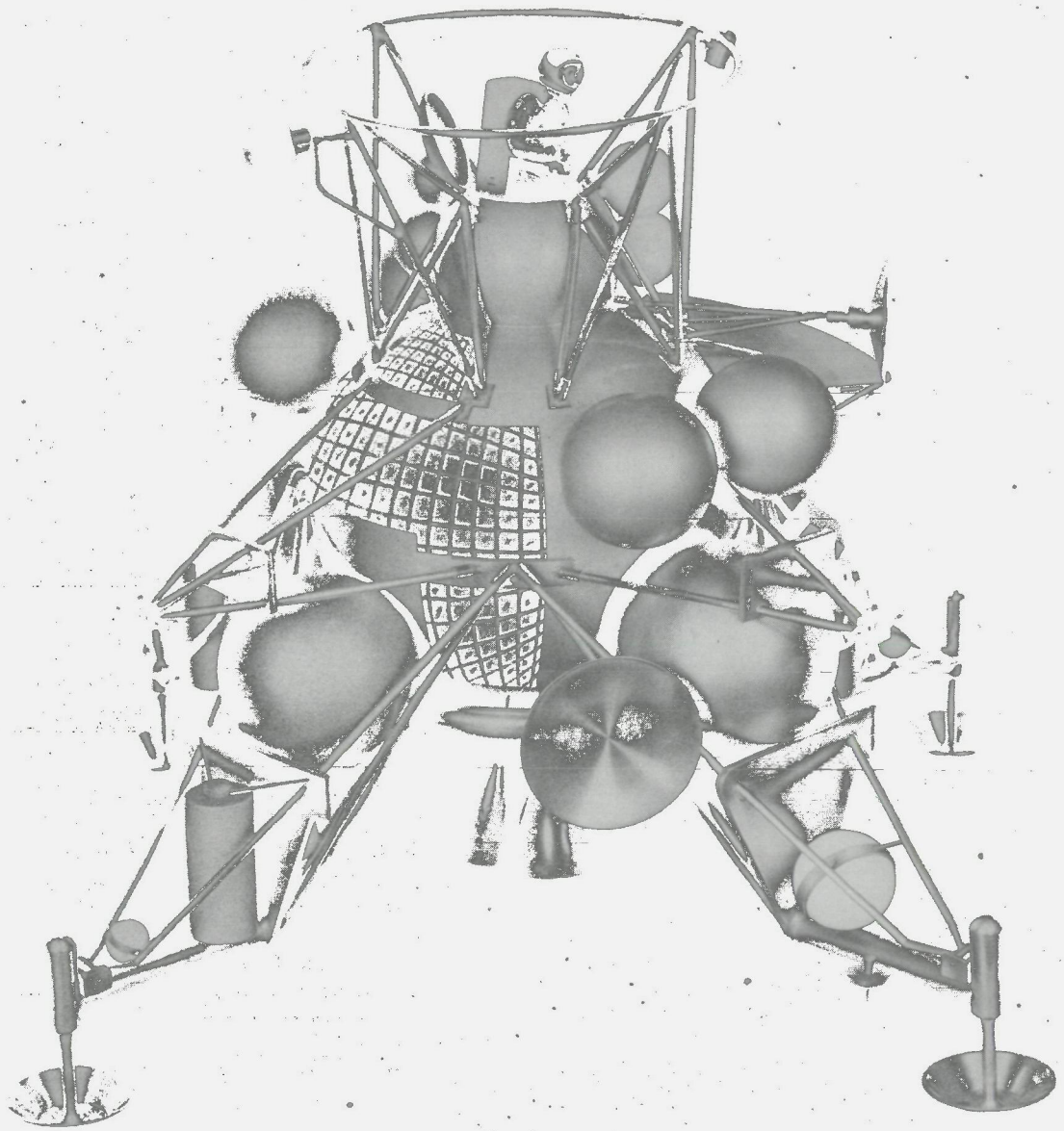


Figure 16–40 RECOMMENDED ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC

CONFIDENTIAL

CONFIDENTIAL



LUNAR EXCURSION MODULE